GPO PRICE $ ________

CFSTI PRICE(S) $ ________

Hard copy (HC) 7.76

Microfiche (MF) 2.25

ff 653 July 65

NASA CR 71503

D2-82309-1

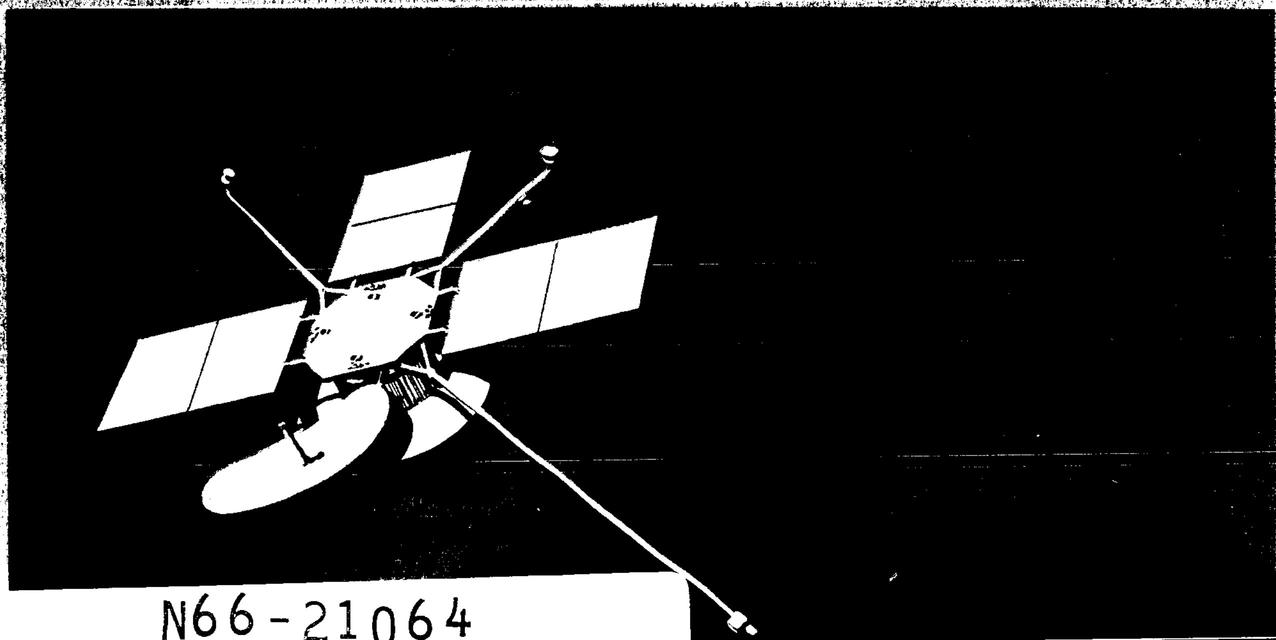
PART I

SPACECRAFT SYSTEM

FINAL TECHNICAL REPORT

VOLUME A

P[illegible]REFERRED DESIGN FOR FLIGHT SPACECRAFT AND HARDWARE SUBSYSTEMS



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N66-21064	
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prepared for **JET PROPULSION LABORATORY**
CALIFORNIA INSTITUTE OF TECHNOLOGY

UNDER CONTRACT NO. 951111 JULY 1965

NAS-7-100

THE BOEING COMPANY • AERO-SPACE DIVISION • SEATTLE, WASHINGTON

NASA CR71503

VOYAGER SPACECRAFT SYSTEM

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PASADENA, CALIFORNIA

UNDER
CONTRACT NO. 951111 NAS-7-100
JULY 1965

THE BOEING COMPANY • AERO-SPACE DIVISION • SEATTLE, WASHINGTON

THE BOEING COMPANY

SEATTLE, WASHINGTON 98124

LYSLE A. WOOD
VICE PRESIDENT-GENERAL MANAGER
AERO-SPACE DIVISION

July 29, 1965

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California

Gentlemen:

This technical report culminates nearly three years of Mariner/Voyager studies at Boeing. During this time, we have gained an appreciation of the magnitude of the task, and feel confident that the experience, resources and dedication of The Boeing Voyager Team can adequately meet the challenge.

The Voyager management task is accentuated by three prime requirements: An inflexible schedule of launch opportunities; the need for an information-retrieval system capable of reliable high-traffic transmission over interplanetary distances; and a spacecraft design flexible enough to accommodate a number of different mission requirements. We believe the technical approach presented here satisfies these design requirements, and that management techniques developed by Boeing for space programs will assure delivery of operable systems at each critical launch date.

Mr. E. G. Czarnecki has been assigned program management responsibility. His group will be ably assisted by Electro-Optical Systems in the area of spacecraft power, Philco Western Development Laboratories will be responsible for telecommunications, and the Autonetics Division, North American Aviation will provide the auto-pilot and attitude reference system. This team has already demonstrated an excellent working relationship during the execution of the Phase IA contract, and will have my full confidence and support during subsequent phases.

This program will report directly to George H. Stoner, Vice President and Assistant Division Manager for Launch and Space Systems. Mr. Stoner has the authority to assign the resources necessary to meet the objectives as specified by JPL.

The Voyager Spacecraft System represents to us more than a business opportunity or a new product objective. We view it as a chance to extend scientific knowledge of the universe while simultaneously contributing to national prestige and we naturally look forward to the opportunity of sharing in this adventure.

Lysle A. Wood

Lysle A. Wood

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INTRODUCTION

INTRODUCTION

In fulfillment of the Jet Propulsion Laboratory (JPL) Contract 951111, the Aero-Space Division of The Boeing Company submits the Voyager Spacecraft Final Technical Report. The complete report, responsive to the documentation requirements specified in the Statement of Work, consists of the five following documents:

VOLUME	TITLE	BOEING DOCUMENT NUMBER
A	Preferred Design Flight Spacecraft and Hardware Subsystems	D2-82709-1
	Part I	
	Section 1.0 Voyager 1971 Mission Objectives and Design Criteria	
	Section 2.0 Design Characteristics and Restraints	
	Section 3.0 System Level Functional Descriptions of Flight Spacecraft	
	Part II	
	Section 4.0 Functional Description for Spacecraft Hardware Subsystems	
	Part III	
	Section 5.0 Schedule and Implementation Plan	
	Section 6.0 System Reliability Summary	
	Section 7.0 Integrated Test Plan Development	
B	Alternate Designs Considered--Flight Spacecraft and Hardware Subsystems	D2-82709-2
C	Design for Operational Support Equipment	D2-82709-3
D	Design for 1969 Test Spacecraft	D2-82709-4
E	Design for Operational Support Equipment for 1969 Test Flight Spacecraft	D2-82709-5

For convenience the highlights of the above documentation have been summarized to give an overview of the scope and depth of the technical effort and management implementation plans produced during Phase IA. This summary is contained in Volume 0, Program Highlights and Management Philosophy, D2-82709-0. A number of supporting documents are provided to furnish detailed information developed through the course of the contract and to provide substantiating reference material which would not otherwise be readily available to JPL personnel. Additionally, a full scale mockup of the preferred design spacecraft has been assembled. This mockup, shown in Figure 1, has been delivered to JPL. The mockup has been provided with the view that it would be of value to JPL in subsequent Voyager Spacecraft System planning. Mr. William M. Allen, President of The Boeing Company, Mr. Lysle A. Wood, Vice-President and Aero-Space Division General Manager, Mr. George H. Stoner, Vice-President and Assistant Division Manager responsible for Launch and Space Systems activities, and Mr. Edwin G. Czarnecki, Voyager Program Manager, are shown with the mockup.

During the 3-month period covered by Contract 951111, Boeing has:

1) Performed system analysis and trade studies necessary to achieve an optimum or preferred design of the Flight Spacecraft.
2) Determined the requirements and constraints which are imposed upon the Flight Spacecraft by the 1971 mission and by the other systems and elements of the project, including the science payload.
3) Developed functional descriptions for the Flight Spacecraft and for each of its hardware subsystems, excluding the science payload.

Figure 1: Preferred Design Mockup

Left to Right:

William M. Allen
Edwin G. Czarnecki
Lysle A. Wood
George H. Stoner

4) Determined the requirements for the Flight Spacecraft associated Operational Support Equipment (OSE) necessary to accomplish the Voyager 1971 mission.

5) Developed a preliminary design of the OSE.

6) Developed functional descriptions for the OSE.

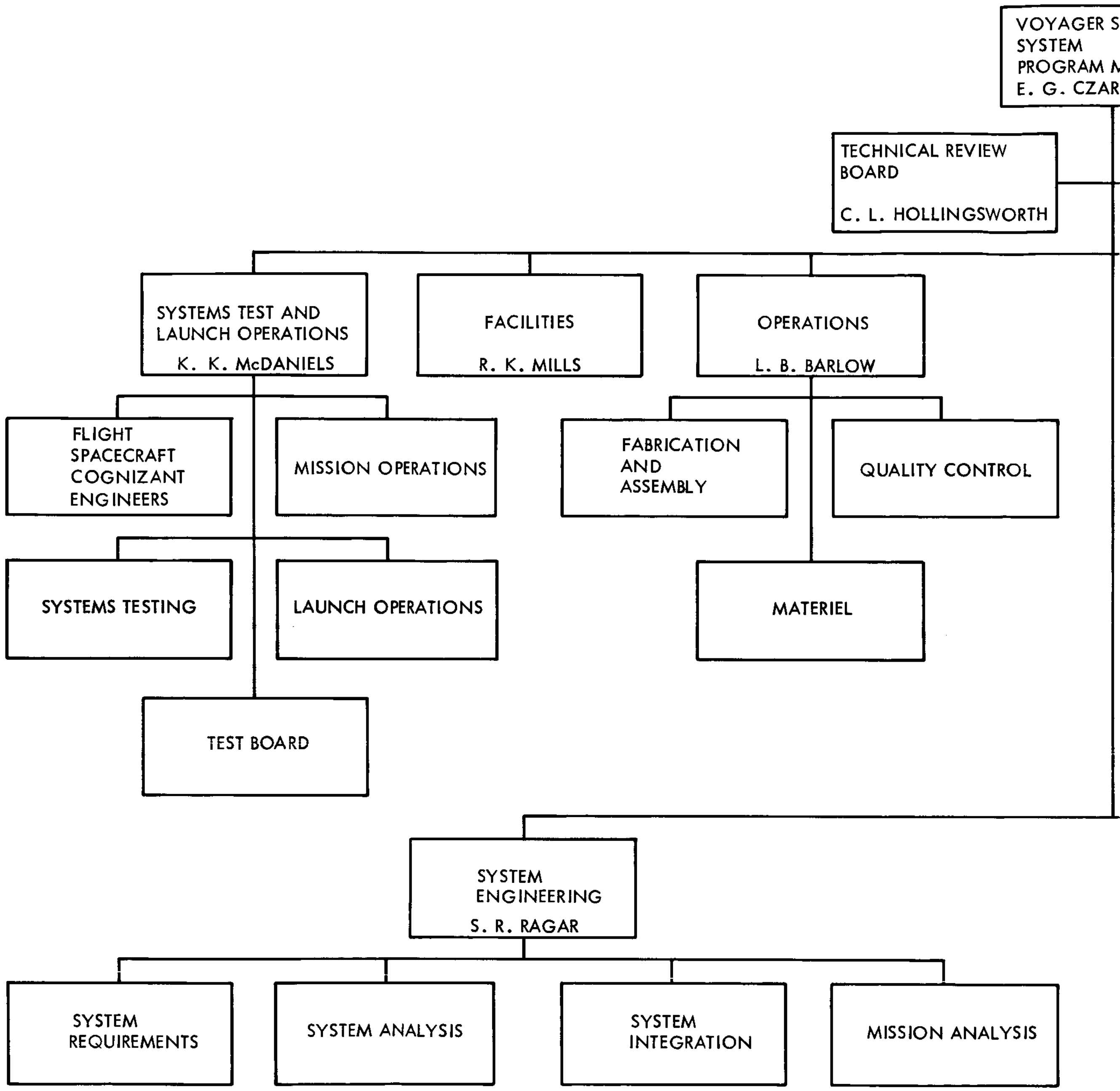
7) Determined the objectives of a 1969 test flight and the design of the 1969 Test Flight Spacecraft using the Atlas/Centaur Launch Vehicle. An alternate test flight program is presented which utilizes the Saturn 1B/Centaur Launch Vehicle.

8) Deveoped functional descriptions for the Flight Spacecraft Bus, and its hardware subsystems, and OSE for the 1969 test spacecraft.

9) Updated and supplemented the Voyager Implementation Plan originally contained in the response to JPL Request for Proposal 3601.

The Voyager program management Team, shown in Figure 2 is under the direction of Mr. Edwin G. Czarnecki. Mr. Czarnecki is the single executive responsible to JPL and Boeing management for the accomplishment of the Voyager Spacecraft Phase IA, and will direct subsequent phases of the program. He reports directly to Mr. George H. Stoner who has the authority to commit those corporate resources necessary to fulfill JPL's Voyager Spacecraft System objectives.

Although Boeing has a technical management capability in all aspects of the Voyager Program, it is planned to extend this capability in depth through association with companies recognized as specialists in certain fields. Use of team members to strengthen Boeing's capability was considered early during pre-proposal activities. The basic concept


VOYAGER S
SYSTEM
PROGRAM M
E. G. CZAR
TECHNICAL REVIEW BOARD
C. L. HOLLINGSWORTH
SYSTEMS TEST AND LAUNCH OPERATIONS
K. K. McDANIELS
FACILITIES
R. K. MILLS
OPERATIONS
L. B. BARLOW
FLIGHT SPACECRAFT COGNIZANT ENGINEERS
MISSION OPERATIONS
FABRICATION AND ASSEMBLY
QUALITY CONTROL
SYSTEMS TESTING
LAUNCH OPERATIONS
MATERIEL
TEST BOARD
SYSTEM ENGINEERING
S. R. RAGAR
SYSTEM REQUIREMENTS
SYSTEM ANALYSIS
SYSTEM INTEGRATION
MISSION ANALYSIS

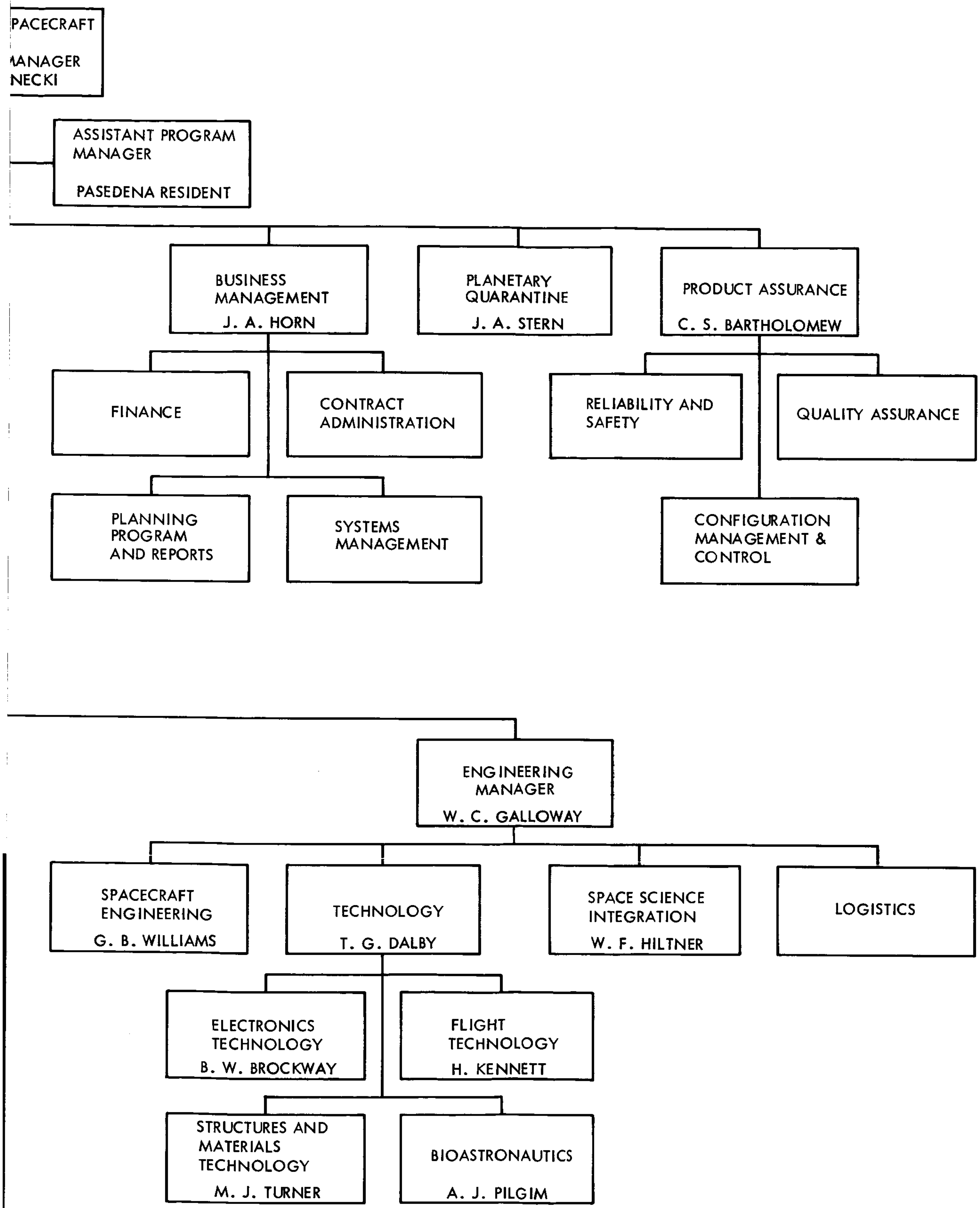


Figure 2 Boeing Voyager
Spacecraft Systems Management Structure

was to add team members who would complement Boeing experience and capability, and significantly improve the amount and quality of technical and management activities. Based upon competitive considerations including experience and past performance and giving strongest emphasis to technical qualifications and management willingness to support the Voyager effort, Autonetics, Philco Western Deveopment Laboratories, and Electro-Optics Systems were chosen as team members. This team arrangement, subject to JPL approval, is shown in Figure 3. The flight spacecraft design and integration task to be accomplished by this team is illustrated in Figure 4. Discussions leading to the formation of this team were initiated late in 1964, formal work statement agreements have been arrived at, and there has been a continuous and complete free exchange of information and documentation; permitting the Boeing team to satisfy JPL's requirements in depth and with confidence.

<table>
<tr><td colspan="3">BOEING VOYAGER TEAM

VOYAGER SPACECRAFT AND SPACE SCIENCES PAYLOAD INTEGRATION CONTRACTOR

The Boeing Company
Seattle, Washington

Mr. E. G. Czarnecki - Program Manager</td></tr>
<tr><td>SUBCONTRACTOR

Autonetics, North American Aviation
Anaheim, California

Autopilot and Attitude Reference Subsystem

Mr. R. R. Mueller
Program Manager</td><td>SUBCONTRACTOR

Philco, Western Development Laboratories
Palo Alto, California

Telecommunications Subsystem

Mr. G. C. Moore
Program Manager</td><td>SUBCONTRACTOR

Electro-Optical Systems Incorporated
Pasadena, California

Electrical Power Subsystem

Mr. C. I. Cummings
Program Manager</td></tr>
</table>

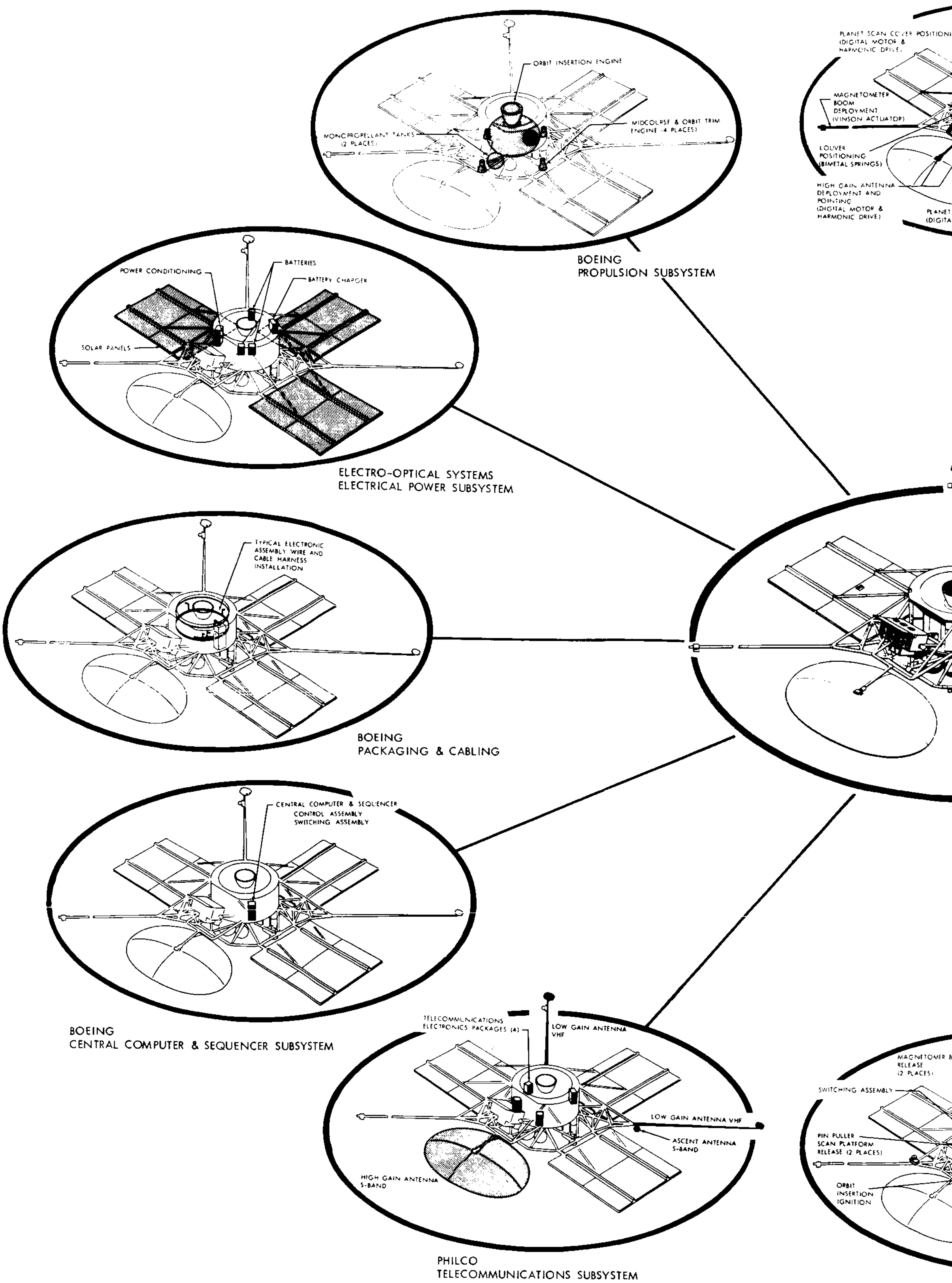
Figure 3

SUMMARY--VOLUME A

The Boeing team's flight spacecraft represents a conservative design based upon selection of space-proven components. The design meets the objectives of the Voyager program for 1969 through 1977 opportunities. The 250-pound science payload, as well as the 2300 or 4500 pound flight capsule can be accommodated and all program and mission objectives achieved.

The Voyager Spacecraft is shown in Figure 4 with equipment deployed in the operational configuration. It is 30 feet wide from solar panel tip to solar panel tip, and the body is 59-inches high. The 31-foot magnetometer boom and 17- and 18-foot antenna booms are shown in position. Estimated weight at this state of the preliminary design is 1565 pounds for the spacecraft, and 3400 pounds for the propulsion module. A contingency of 285 pounds of the specification weight of 5250 pounds is available for selective use during the detail design phase. The 20 equipment modules are fastened to the central magnesium shell with cooling provided by thermal radiation from the external faces of the package. Thermal control is by space-facing louvers.

Outstanding design features of the Boeing team's Voyager Spacecraft are its ability to perform reliably, transmit data to Earth at encounter at the 50,000 bit-per-second rate generated in the science package, and meet all mission energy requirements through 1977 with a single propulsion module design. Use of redundancy in critical components and selection of proven designs requiring a minimum of additional development


ORBIT INSERTION ENGINE
MONOPROPELLANT TANKS (2 PLACES)
MIDCOURSE & ORBIT TRIM ENGINE (4 PLACES)
BOEING
PROPULSION SUBSYSTEM
PLANET SCAN COVER POSITIONI
(DIGITAL MOTOR & HARMONIC DRIVE)
MAGNETOMETER BOOM DEPLOYMENT (VINSON ACTUATOR)
LOUVER POSITIONING (BIMETAL SPRINGS)
HIGH GAIN ANTENNA DEPLOYMENT AND POINTING (DIGITAL MOTOR & HARMONIC DRIVE)
PLANET
(DIGITA
BATTERIES
POWER CONDITIONING
BATTERY CHARGER
SOLAR PANELS
ELECTRO-OPTICAL SYSTEMS
ELECTRICAL POWER SUBSYSTEM
TYPICAL ELECTRONIC ASSEMBLY WIRE AND CABLE HARNESS INSTALLATION
BOEING
PACKAGING & CABLING
CENTRAL COMPUTER & SEQUENCER
CONTROL ASSEMBLY
SWITCHING ASSEMBLY
BOEING
CENTRAL COMPUTER & SEQUENCER SUBSYSTEM
TELECOMMUNICATIONS ELECTRONICS PACKAGES (4)
LOW GAIN ANTENNA VHF
LOW GAIN ANTENNA VHF
ASCENT ANTENNA S-BAND
HIGH GAIN ANTENNA S-BAND
PHILCO
TELECOMMUNICATIONS SUBSYSTEM
MAGNETOMER B
RELEASE (2 PLACES)
SWITCHING ASSEMBLY
PIN PULLER SCAN PLATFORM RELEASE (2 PLACES)
ORBIT INSERTION IGNITION

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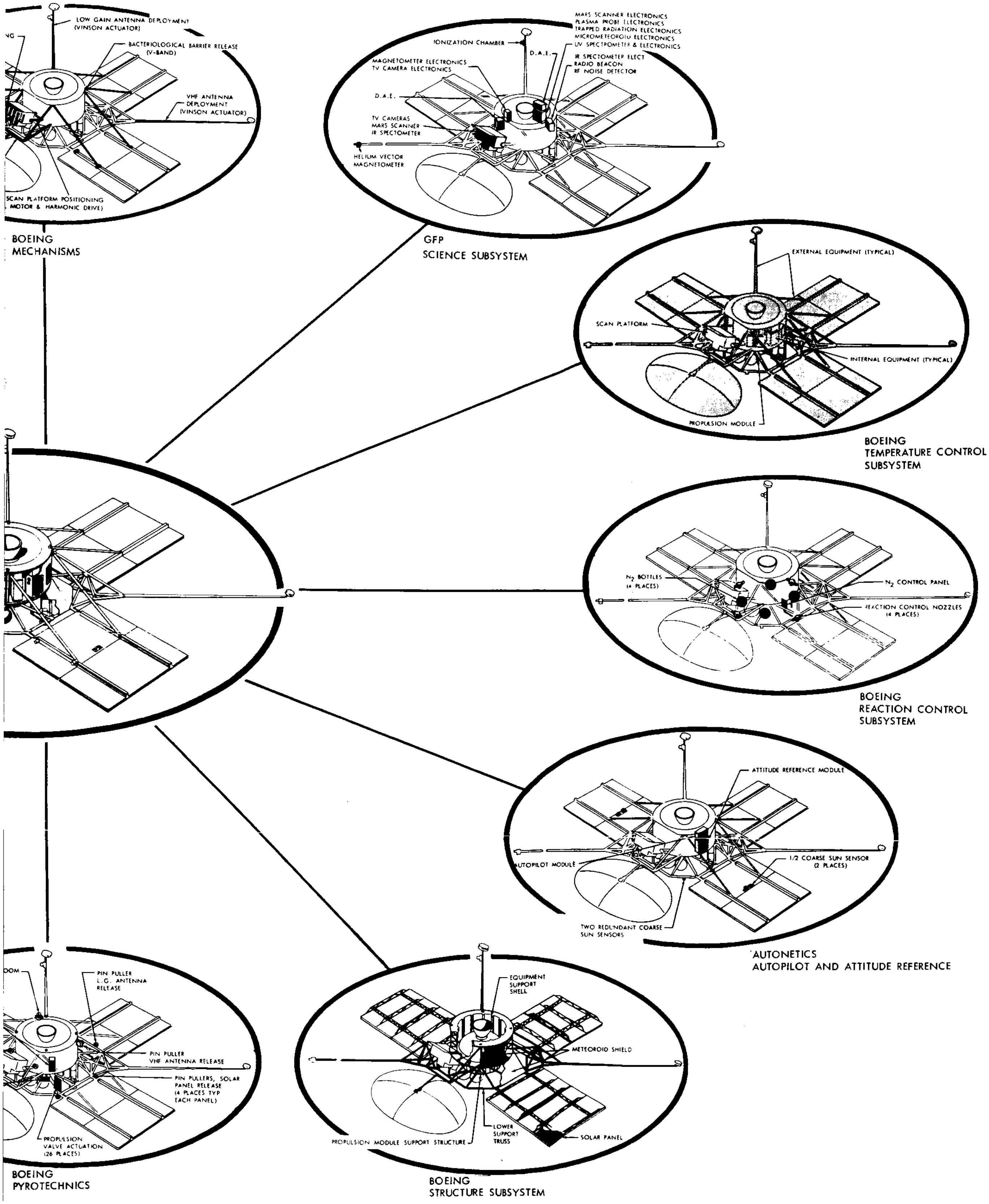


Figure 4. Voyager Flight Spacecraft Subsystem Integration

resulted in an overall mission success probability of 47 percent, exceeding the specified 45 percent, including an allocation of 0.674 for the science payload.

The spacecraft can enter biologically safe orbits with periods as low as 18 hours from Mars approach velocities as high as 3.5 km/sec., or with periods less than 9 hours from approach velocities as high as 3.0 km/sec. The 18-hour orbit provides coverage of four different swaths of Mars surface in the first three days after encounter.

In 1971, orbits are available which have no occultation of Canopus or the Sun for the first 60 days in orbit. The periapsis positions are at southern latitudes and at illumination angles which favor the black and white TV experiment. Some adjustment of periapsis position is available with "off-periapsis" orbit insertion techniques. The "off-periapsis" insertion technique allows the utilization of the fixed-total-impulse solid motor for all approach velocities considered.

The telecommunications design includes completely redundant radio subsystems. It features an 8' x 12' paraboloidal high-gain antenna, two 50-watt traveling wave tubes and bi-orthogonal block coding to obtain the high data rate. The 50-watt tube selection is supported by three separate tube designs including test data. Detailed link calculations substantiate a positive communication link margin under worst-case conditions at Mars encounter, with a calculated 48,000 bits per second data rate. (Upon definition of the precise science payload data rate, the telecommunications link can be optimized to that value.) For longer communication ranges, alternate lower data modes and two tape recorders with storage capability for 2 x 10^8 bits of scientific data are provided. Two 72,000 bit buffers provide temporary storage of spacecraft engineering and capsule data.

The spacecraft propulsion subsystem consists of a solid motor with an oblate spheroidal case for Mars orbit insertion and four 50-pound thrust, jet vane controlled, hydrazine engines operating in pairs for midcourse and orbit trim. The solid propellant motor with a specific impulse of about 300 pounds force seconds per pound mass delivers 10,500 pounds maximum thrust and burns regressively to provide not more than 2.2 g's acceleration. Solid motor TVC is by a Freon secondary injection system. With the available 2306 pounds of solid propellant, an orbit insertion velocity increment of 5700 feet per second is attained. The 50-pound thrust monopropellant engines with a specific impulse of 235 pound force seconds per pound mass have multiple restarting capability. These engines utilize the spontaneous decomposition catalyst. Hydrazine fuel capacity is adequate for 929 total seconds of operation.

Reaction control is produced by expulsion of sterile nitrogen through two redundant sets of eight .25 pound thrusters each, which are body-mounted on the spacecraft. Four titanium tanks contain 60 pounds of cold nitrogen for reaction control and propulsion requirement. The 45 pounds allocated to reaction control is adequate for the 6-month orbital mission with a safety factor of 2. Under nominal conditions, the nitrogen supply is adequate for four years. Both propulsion systems, plus the reaction control subsystem, are assembled in a single sub-module mounted in the spacecraft. This modular arrangement permits complete assembly and checkout, including sterilization, prior to installation on the spacecraft. The propulsion and reaction control systems including all fuel and gas supplies are sterilized to avoid planetary contamination by propulsion ejecta.

The selected attitude reference and autopilot subsystems are comprised of an attitude reference module, autopilot module, and coarse and fine Sun sensors. The attitude reference module includes three redundant Autonetics G-10 gas-bearing gyros, two redundant accelerometers, two redundant Canopus sensors and two fine Sun sensors. The coars Sun sensors are located on two solar panels. The autopilot is an analog type and maintains spacecraft orientation to within $\pm$0.4 degree in cruise, $\pm$0.2 degree in Mars orbit, and the limit cycle period is several hours. All selected components are existing designs with operation and qualification experience.

The electrical power system is similar to Mariner IV, with three solar panels, 8-1/2' x 13', consisting of two sections each. The total area of 236 square feet provides 627 watts of power at the distance of Mars from the Sun. A flat solar cell arrangement is used; three silver cadmium batteries are provided for use during off-Sun periods. The power subsystem regulates and distributes the electrical power to subsystems where additional power conditioning is performed. A 50-percent increase in power is possible by addition of one section to each solar panel.

The Voyager central computer and sequencer (CC&S) provides timing functions and command signals to all other spacecraft subsystems. A magnetic core memory provides storage for 256 21-bit words and a capability to execute 333 different commands. The CC&S minimizes the need for detail ground commands by incorporating preplanned operational sequences. All commands and stored instructions can be monitored and controlled from the ground for complete analysis and control during the entire mission. A modified NASA Lunar Orbiter programmer has been selected as the basic element. This memory-oriented digital computer has been space-qualified and addition of redundant data processing and switching circuits provide a highly reliable unit.

The spacecraft structure includes a simple truss base, 10 feet wide at the bottom and 5 feet wide at the top, fabricated of 6AL4V titanium tubing. This base attaches to the Centaur adapter and supports the antenna and solar panel appendages. The electronic packages are connected to a five-foot diameter, cylindrical, magnesium shell installed above the truss. The flight capsule is supported by an adapter ring with loads carried by four columns through the cylindrical shell.

A number of major technical problems were encountered and studied in developing the preliminary design. The most significant of these were as follows:

1) The assessment of the most reliable and highest power transmitter tube meeting the Voyager requirements;

2) The overall spacecraft magnetics problem with particular attention to the magnetic focusing field for the traveling wave tube.
3) Availability and reliability of spacecraft recorders.
4) Selection of a reliable secondary battery with adequate recycle life.
5) Estimation of solar panel degradation from electromagnetic radiation and meteoroids during the mission.
6) The trade-off between proven instruments versus new and inherently simpler instruments.
7) Determination of the degree and type of redundancy, for example, using two identical instruments of two ~~difference~~ different designs.
8) The effect of the solid engine exhaust on the structure and solar panel temperature.
9) Accommodating the length of the orbit insertion engine.
10) Selection of installation technique for the equipment packages.
11) Selection of the thrust vector control technique.
12) Effect of heat soak sterilization on equipment.

These problems are the key technical considerations in developing the preferred design.

The subsystems of the Boeing team's spacecraft provide a conservative and highly reliable design. No state-of-the-art advances are required to meet the design criteria for any subsystem.

CONTENTS

1.0 VOYAGER 1971 MISSION OBJECTIVES AND DESIGN CRITERIA

1.1 PROGRAM OBJECTIVES

The objectives and design criteria set forth in JPL Document 45, "Preliminary Voyager 1971 Mission Specification," May 1, 1965, and Document 46, "Voyager 1971 Mission Guidelines," May 1, 1965, have been followed without exception and are summarized in Section 1.0 of this volume.

The mission objectives and design criteria imposed on the Flight Spacecraft and its associated subsystems in performance of a flight mission to the planet Mars during the 1971 opportunity are defined in consonance with the following Voyager program objectives.

Primary--The primary objective is an orderly program of continuous improvement of knowledge in science and technology to achieve the following in an efficient, timely manner:

1) Scientific and engineering observations and experiments directed toward extending Voyager Spacecraft System capability to operate near the planet and on the planet surface, and the development of this capability during the life of the program;
2) Scientific and engineering observations and experiments directed toward extending the capability of the scientific instruments to operate near the planet, and on the planet surface; specific definition of future experiments concerning exobiology and planetology; and development of this capability during the life of the program;

3) Scientific observations and experiments concerning possible biology and biochemistry of Mars;

4) Scientific observations and experiments concerning the physics and chemistry of the Martian lithosphere and atmosphere directed toward obtaining information essential to advancement of planetology.

Secondary--The secondary objective is to perform certain field and particle measurements in interplanetary space between the orbits of Earth and Mars.

1.2 1971 MISSION OBJECTIVES

The primary objective of the 1971 mission is to develop and begin the use of the basic capability to place significant payloads at Mars, conduct observations of Martian phenomena over extended periods, and transmit the results of those observations to Earth. The objective is ordered in the following way, with estimates of desired cumulative probabilities of success for each flight stated for each subobjective.

1) Perform a successful launch and injection of the Planetary Vehicle into a prescribed transfer orbit (90 percent probability of success).

2) Perform a successful spacecraft-capsule separation maneuver at a preselected time and location (80 percent probability of success).

3) Place an operating Science Payload in a selected orbit about Mars and perform the functions necessary to begin orbital operations (65 percent probability of success).

4) Perform orbital operations to obtain data from the orbital Science Payload and return the data to Earth for a specified time of 1 month and as long thereafter as possible (45 percent probability of success).

5) Place the Flight Capsule on a selected impact trajectory to Mars (75 percent probability of success).

6) Enter the Mars atmosphere and obtain data on the lower Mars atmosphere from the Capsule Science Payload (65 percent probability of success).

7) Land the Flight Capsule, establish communications with Earth, and return entry, landing, and system status data to Earth (45 percent probability of success).

8) Perform landed operations to obtain data with Capsule Science Payload over at least one Martian diurnal cycle and return the data to Earth (35 percent probability of success).

A secondary objective is to develop experience with both flight and ground systems that are required for delivery and operation of the Spacecraft Science Payload, for ferrying and separation of the capsule, and for delivery and operation of the Capsule Science Payload.

A tertiary objective is to make scientific and engineering observations in interplanetary space during the transit flight from Earth to Mars and transmit the resulting data back to Earth.

A quaternary objective is to provide specific flight and ground designs and equipment elements that will be compatible with subsequent Voyager missions to Mars.

1.2.1 Flight Spacecraft Objectives

The specific objectives of the Flight Spacecraft are to deliver and operate the Spacecraft Science Payload and to ferry and separate the Flight Capsule.

The mission objectives of the Flight Spacecraft relative to scientific measurements are to:

1) Search for the location and characteristics of life on Mars by use of imaging and radiometric techniques;
2) Measure Mars atmospheric constituents and characteristics, including refractive index variations, temperature profile, pressure profile, Faraday rotation, and back and forward scattering;
3) Conduct detailed pictorial exploration of Mars, including (a) definition of major geographical features; (b) determination of geological characteristics in selected areas; (c) determination of color content and infrared radiation patterns at selected areas; and (d) determination of meteorological phenomena such as cloud structure, circulation patterns, and fog belts;
4) Measure the particle and field characteristics in the vicinity of Mars;
5) Improve knowledge of Mars gravitational field;
6) Measure albedo and phase characteristics of the planet at selected wavelengths;

7) Improve knowledge of Mars ephemeris;

8) Measure interplanetary particle and field characteristics, including solar plasma, solar and galactic magnetic fields, particle events, and micrometeoroid flux.

1.2.2 Flight Capsule Objectives

Although the 1971 mission objectives are oriented principally toward the Flight Spacecraft, the Flight Capsule experimental objectives are included to facilitate the investigation of interfaces and interactions. The mission objectives of the Flight Capsule are to:

1) Make life detection measurements;

2) Measure the atmospheric characteristics in the Mars entry zone, including composition profile and density profile;

3) Measure near-surface atmosphere and meteorological characteristics, including density, temperature, light scattering, and wind velocity;

4) Conduct a pictorial survey in the vicinity of the landing site.

1.3 MISSION RESTRAINTS

Two Saturn IB/Shrouded Centaur Launch Vehicles will be provided for the 1971 opportunity. Launch Pads 34 and 37B at ETR will be used to permit a launch separation interval as short as 2 days. The capability to launch within a designated 1-hour period is required.

A nominal launch opportunity of 50 days will be provided with a minimum daily firing window of 2 hours.

The design of the spacecraft will be compatible with the use of 85-foot and 210-foot antennas operating at S-band. Three 85-foot antennas will be employed from near spacecraft injection until near encounter with Mars and the 210-foot antennas will be employed for the orbiting operations. Three 85-foot antenna station nets will be used for coverage from near spacecraft injection to near planetary encounter. The specific stations will be selected from the following complexes: Johannesburg, South Africa; Madrid, Spain; Canberra and Woomera, Australia; and Goldstone, California. The Goldstone, Canberra, and Madrid 210-foot antenna will be used for coverage during the later phases of transit and during the orbital and landed operations in the vicinity of, and on, Mars.

Planetary Quarantine--The probability that Mars will be contaminated prior to Calendar Year 2021 as a result of any single launch shall be not greater than 1 in 10,000. Consideration will be given to the implications of this requirement for the Centaur stage, the spacecraft, the capsule, and all emissions and ejecta.

1.4 DESIGN CRITERIA

1.4.1 Design Approach

The design approach will give precedence to reliability, with performance, schedule, maintainability, and cost receiving major attention.

Reliability will be ensured through conservative design, space-proven hardware, environmental control, rigorous qualification of all hardware not space-proven, modular assemblies, test procedures designed to isolate failure modes, and redundancy. Based on present state of the art for Voyager, redundancy will be required to meet the 1971 Mission Success Criteria.

1.4.2 Design Considerations

Structural design criteria correspond with those given in MC-4-521-A, "Mariner C Flight Equipment Structural Design Criteria."

1.4.2.1 Margin of Safety

The margin of safety shall be considered at both yield and ultimate load levels. Margin of safety shall be defined as:

$$MS = \frac{\text{allowable load (or allowable stress)}}{\text{design load (or design stress)}} - 1$$

1.4.2.2 Hazard Factor

The hazard factors to be used in the design of the Flight Spacecraft, or any component thereof, are presented in the following table.

ITEM	HAZARD FACTOR
Pressure vessels hazardous to personnel in the event of failure	1.76
All other components, including pressure vessels, nonhazardous to personnel in the event of failure	1.0

1.5 PLANETARY VEHICLE WEIGHT

Weights for the Planetary Vehicle for the 1971 mission are allocated as follows:

	JPL SPECIFICATION WEIGHT	
Flight Capsule separated weight	1950	
Flight Capsule adapter and sterilization canister (a maximum of 150 pounds may remain with spacecraft)	350	
Flight Spacecraft minus spacecraft adapter (includes 250 pounds of science)	5500	
Planetary Vehicle separated weight		7800
Spacecraft adapter and spacecraft support above field joint	250	
Planetary Vehicle weight		8050
Spacecraft support below field joint	250	
TOTAL		8300

1.6 COMPETING CHARACTERISTICS

When there are conflicting technical requirements, the following order of priority relative to acceptable risks shall govern:

1) Meeting the requirement for planetary quarantine.

2) Proper operation of telemetry and communication equipment in down link.

3) Continuous, proper Sun-line attitude orientation of spacecraft.

4) Continuous, proper temperature control of spacecraft.

5) Proper functioning of power equipment on spacecraft.

6) Proper operation of communications and command equipment (uplink).

7) Proper roll control of spacecraft.

8) Proper exectuion of midcourse maneuvers.

9) Proper spacecraft capsule separation.

10) Proper execution of the manuever placing the spacecraft in a useful Mars orbit.

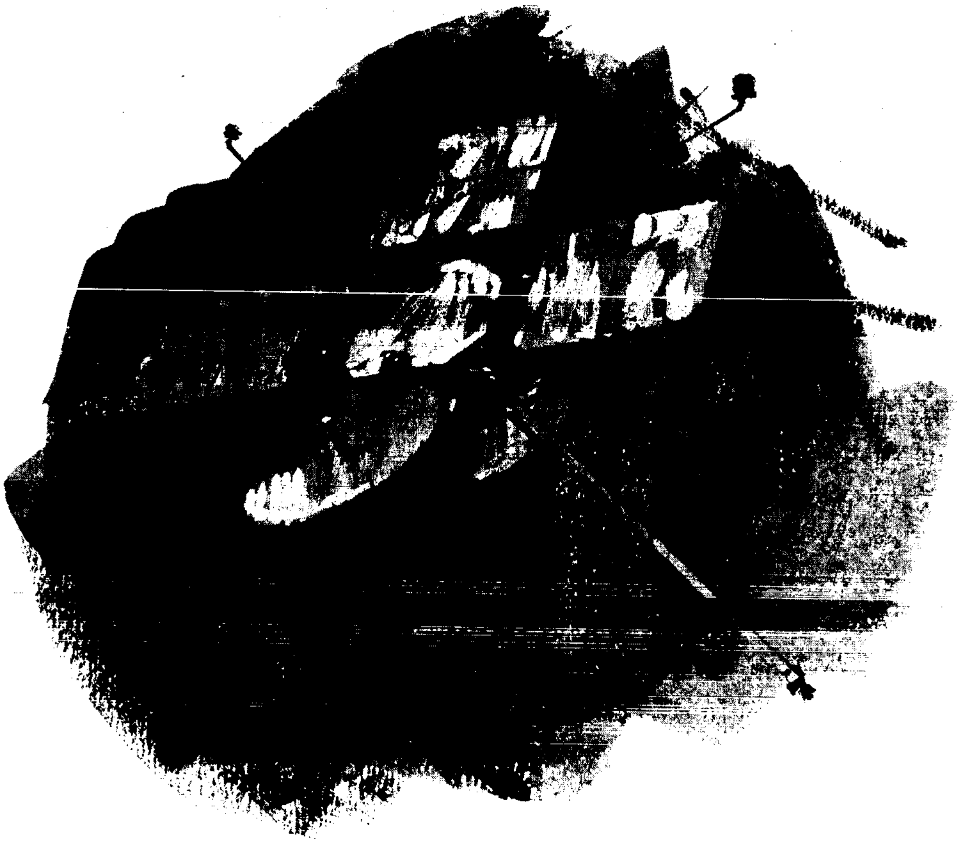
11) Proper operation of spacecraft instrumentation at Mars.

12) Proper execution of the maneuver placing the capuule on a useful Mars landing trajectory.

13) Proper operation of the cruise instrumentation.

14) Design value to the 1973 and subsequent missions.

15) Equipment applicability to the 1973 flight hardware and ground hardware.



CONTENTS

2.0 DESIGN CHARACTERISTICS AND RESTRAINTS

The design characteristics and restraints as set forth by JPL in Document Number 45, Preliminary Voyager 1971 Mission Specification, May 1, 1965, and Document Number 46, Voyager 1971 Mission Guidelines, May 1, 1965, have been followed without exception and are summarized in Section 2 of this volume. Design values for the Mars magnetically trapped radiation and the Mars meteoroid environment were selected recognizing the "upper limit" for the radiation and "worst case" for the meteoroid environments mentioned in Document Number 45. Justification for the selected design values is given in the appropriate subsections.

2.1 DESIGN CHARACTERISTICS

2.1.1 General

This subsection describes the design characteristics of the Flight Spacecraft.

The launch vehicle shall satisfy the following requirements imposed by the Planetary Vehicle:

1) The launch vehicle shall launch the Planetary Vehicle on a trajectory that results in a Mars encounter;
2) The launch vehicle shall protect the Planetary Vehicle from the aerodynamic environment;
3) The launch vehicle shall relay certain telemetry data from the Planetary Vehicle during the boost phase;
4) The launch vehicle shall separate from the Planetary Vehicle after burnout of the third stage;
5) The launch-vehicle system shall provide active environmental control required by the Planetary Vehicle and including:
   a) Cooling and heating requirements on the pad during test and prior to launch;
   b) Cleanliness requirements when encapsulated under the nose fairing;
   c) Limits on heating of the Planetary Vehicle by the nose fairing during boost;
   d) Humidity control of the atmosphere under the nose fairing;
   e) Minimum altitude for nose fairing separation;
   f) Venting requirements during ascent.
   g) Dynamic environment of the Planetary Vehicle during launch.

On-board sequencing and logic, as well as ground-command capability, shall be provided. The spacecraft will provide such services to the Flight Capsule as power, timing and sequencing, telemetry, and command during the transit portion of the missions. It shall have a two-way communication system which provides telemetry to Earth and command capability.

2.1.2 Mission Profile

2.1.2.1 Boost Phase

From liftoff until shroud ejection, the Flight-Spacecraft telemetry signals will be radiated through a parasitic antenna located on the shroud. After shroud ejection, communications will be maintained via the Flight-Spacecraft ~~low gain~~ ascent antenna.

A parking-orbit ascent mode shall be used for the Mars 1971 mission. An arbitrary limit for a 25-minute parking orbit exists for the 1971 mission; the minimum parking-orbit coast time shall be 2 minutes.

The launch vehicle will inject the Planetary Vehicle on a trans-Mars trajectory and will provide the signal to initiate separation of the spacecraft from the Centaur stage.

Miss plus time-of-flight dispersions of the Planetary Vehicle produced by the Launch vehicle shall be correctable with a maximum midcourse velocity increment of 15 meters per second applied 2 days after injection.

2.1.2.2 Initial Acquisition

After separation of the Planetary Vehicle from the Centaur stage, the Centaur stage shall back away by employing retrothrust. Immediately after separation, solar panels and high- and low-gain antennas will be deployed, and the acquisition of celestial references will commence. The Planetary Vehicle will automatically rotate in pitch and yaw to acquire a solar-reference fix and then be programmed through roll for acquisition of Canopus. Power during acquisition will be supplied by batteries. Solar acquisition will nominally be completed within 90 minutes after injection.

2.1.2.3 Cruise Phase

During the cruise phase, the Planetary Vehicle will remain attitude stabilized. The separation and deflection of the Flight Capsule from the Flight Spacecraft shall not result in loss of the Flight Spacecraft attitude-control references. Continuous operational coverage for both Planetary Vehicles during the cruise phase will be supplied by the Deep Space Network.

The Flight Spacecraft, prior to Flight Spacecraft-Flight Capsule separation (except during maneuvers) shall be capable of accepting data at the rate of 10 bits per second from the Flight Capsule and transmitting the data to Earth, compatible with spacecraft engineering telemetry. The transmitted data will consist of commutated engineering-data frames alternated with science-data frames. The Flight Spacecraft shall have the capability of transferring at least five commands to the Flight Capsule before separation. Those commands may be Flight-

Spacecraft stored commands or ground commands transmitted through the Flight Spacecraft.

2.1.2.4 Midcourse Maneuvers

The Planetary Vehicle shall have the capability to perform at least four midcourse corrections. A sufficient total-midcourse-velocity allocation shall be provided to correct trajectory dispersions and perform any required trajectory biasing (to satisfy the planetary quarantine constraint) with a probability of 0.99. The required allocation will be approximately 75 meters per second. The first midcourse maneuver will occur as early as 2 days after launch. Velocity increments for trajectory corrections will be executed through stored command under the control of the guidance subsystem.

2.1.2.5 Flight Capsule - Flight Spacecraft Separation

Flight-Capsule separation will be under control of the capsule. The Flight Capsule shall be mounted forward of the Flight Spacecraft and shall interface with the Flight Spacecraft at the field joint between the two vehicles. The flight-separation joint is contained within the capsule adapter. The flight-separation system is forward of the field joint and is a part of the capsule system.

2.1.2.6 Encounter Phase

Two Flight Spacecraft will be inserted into Mars orbit and shall arrive with a minimum separation of 10 days.

2.1.2.7 Transfer Trajectory

Type-I-transfer trajectories shall be used for the 1971 mission. A maximum C_3 of 18 km^2/sec^2 shall be assumed. This is compatible with providing an adequate mission-weight margin for a separated Planetary-Vehicle weight of 7800 pounds. The hyperbolic excess velocity at Mars shall not exceed a maximum of 5 Kilometers per second. In order to improve orbit redetermination geometrics, the absolute value of the declination of the departure asymptote (DLA) shall be greater than 5 degrees and that of the inclination (INC) of the heliocentric transfer plane to the ecliptic plane shall be greater than 0.1 degree. For the 1971 mission, launch from AFETR shall be provided along azimuths ranging from 71 to 108 degrees east of north inclusive. A future expansion of the azimuth sector to include azimuths from 45 to 114 degrees east of north shall be considered. Limiting launch-azimuth boundaries require that DLA be less than or equal to 50 degrees. For preliminary launch azimuth sector planning purposes, the |DLA| shall be taken as less than or equal to 33 degrees for the 1971 mission.

Trajectory corrections for midcourse maneuver and planetary injection will be based upon angular measurements, two-way doppler frequency shift, and range.

2.1.2.8 Approach Characteristics

Orbit insertion and capsule entry, descent, and landing shall occur in view of the DSIF at Goldstone, California.

The selection of an aiming point shall include consideration of landing-point constraints, planetary-quarantine constraints, accuracy requirements, orbit-determination uncertainties, and midcourse maneuvers. The selection of aiming points during an actual flight will follow an adaptive policy.

2.1.3 Subsystems

2.1.3.1 Telecommunications

The telecommunications system shall provide the capability of:

1) Determining the angular position, the doppler frequency shift, and the range of the spacecraft for orbit determination;
2) Receiving transmitted commands from Earth for controlling spacecraft operation;
3) Telemetering engineering and scientific information from the spacecraft.

Planetary-Vehicle and MOS Interfaces--The Planetary-Vehicle and MOS interfaces are in the areas of telecommunications, control data handling, and operating modes. Specific interface areas include:

1) Telecommunications frequencies;
2) Tracking modes;
3) Telemetry subsystem operating modes, data rates, and data formats;
4) Command-subsystem data rates, command formats, and number of commands;
5) On-board-controller command sequences;
6) Operating modes of spacecraft, capsule, and Science Payloads;

7) Equipment operating-tolerance specifications for spacecraft, capsule, and Science Payloads.

Planetary Vehicle and DSN Interface--The Planetary-Vehicle and DSN Interface includes S-band rf links for acquisition of spacecraft telemetry data, sending of spacecraft commands, and spacecraft tracking.

1) The telecommunication equipment shall be compatible with nonmission-oriented Deep Space Instrumentation Facility equipment;
2) The bit error probability for the telemetry link at threshold shall be less than 5×10^{-3};
3) An emergency mode shall be incorporated with degraded performance relative to the specified bit-error rate ($P_e^b = 1 \times 10^{-2}$ instead of 5×10^{-3}). This mode provides engineering data via the spacecraft low-gain antenna for a period of 1 to 3½ months beyond encounter if communications capability using the high-gain antenna is lost. (e.g. because of degraded attitude control);
4) The bit error probability for the command link at threshold shall be less than $P_e^b = 10^{-5}$;
5) The command equipment shall be compatible with the command-verification equipment;
6) The command equipment shall employ pseudo-noise-synchronization techniques.

Spacecraft and Capsule RF-Relay Link--The Flight Spacecraft design shall provide a VHF relay receiver and a fixed low-gain antenna. This equipment shall be capable of receiving post separation data from the capsule (until impact) at a rate of 10 bits per second. This data will be handled and retransmitted to Earth by the spacecraft-communication system. Relay communications after impact shall also be considered.

Antenna Subsystem--The antenna subsystem shall consist of the following:

1) An 8'by 12' paraboloidal high-gain antenna, capable of maintaining continuous Earth bearing during later stages of spacecraft flight.
2) A low-gain S-band antenna capable of providing uniform coverage for telemetry during the cruise portion ~~A low-gain S-band antenna capable of providing uniform coverage during the near-Earth~~ portion of the flight and command up-link during all phases plus maneuvers;
3) A VHF fixed low-gain antenna and a VHF receiver will be provided to receive data from the capsule after separation (up to and including impact) at a rate of 10 bits per second.

The ground-link antennas will communicate with the DSIF spacecraft-monitor station at Cape Kennedy for spacecraft capsule DSN compatibility verification and telemetry reception from liftoff until end-of-view on the local station horizon.

Receiving Subsystem--During much of the Mars transit time, the Flight Spacecraft will remain attitude-stabilized and will maintain continuous transmission. The transmitted information will consist of commutated engineering-data frames alternated with science-data frames.

The spacecraft engineering-data rate shall be 10 bits per second (unless otherwise specified) and the capsule-to-spacecraft data rate after landing shall be 100 bits per second to allow for multiplexing of engineering and science data as a backup link for the direct capsule-to-Earth link.

Command and Control--The spacecraft-control system shall be designed to conserve the number and complexity of Earth-based commands. The number of real-time Earth-based commands shall be minimized. Spacecraft and capsule shall contain on-board controllers to initiate control commands that are entered prior to the time of execution. The spacecraft shall receive, decode, and execute commands received via Earth-based radio including:

1) Discrete commands to perform functions where execution times cannot be defined prior to launch;
2) Backup commands for critical and other selected on-board controller commands;
3) Quantitative commands associated with spacecraft maneuvers;
4) Commands to override or inhibit critical on-board controller commands or previously received radio commands.

Critical Radio Command Integrity--The Flight Spacecraft design shall provide for telemetry readback or other equally reliable methods for verifying critical radio commands prior to execution. This requirement involves the CC&S as well as telecommunications.

At preselected times throughout the trajectory, several discrete events will occur:

1) The star tracker cone angle will be updated;
2) The bit rate of the transmitted signal will be changed as required;
3) The transmitter output will be switched from the low-gain antenna to the high-gain antenna;

4) The high-gain antenna position will be updated as required.
These events will be initiated by on-board logic with ground-command backups.

Transmitting System--The output from the transmitting subsystem shall possess the capability for connection to either the high-gain or low-gain antenna. Switchover will occur on command from the CC&S with ground-command backup. Transmission via the high-gain antenna will be required for approximately the last half of the mission. The spacecraft will transmit continuously during the transit time to Mars. The transmitted data will be a combination of engineering and science data.

Bit Rates--Science instruments shall possess individual output data rates of 50,000 bits per second for planetary science and 100 bits per second for cruise science. An individual operation cycle of spectrometric, radiometric, and photometric instruments will generate from 10^4 to 10^6 bits, and photographic instruments will generate from 10^6 to 10^7 bits per picture.

Storage--cruise-science data will not be stored because (1) real-time transmission is available during the entire mission (with the possible exception of maneuver, separation, and orbit-insertion periods), and (2) some buffering is assumed in the DAS to handle solar-flare conditions. Planetary science data will be stored as required.

Ranging--The spacecraft shall be instrumented to provide ranging capability. ~~two-way doppler and telemetry data for ranging.~~

2.1.3.2 Central Computer and Sequencer

The CC&S functions are:

1) Generate timing frequencies and pulse trains on the Flight Spacecraft;
2) Provide a timing capability for specific fixed time intervals and events;
3) Provide a timing capability for specific commanded time intervals;
4) Provide control signals based on 1), 2), and 3);
5) Detect commands that are the output of the Flight Spacecraft radio demodulator;
6) Decode the digital commands and route discrete and quantitative commands to spacecraft subsystems;
7) On-board sequences shall include:
    a) Prelaunch;
    b) Launch;
    c) Automatic spatial reference acquisition;
    d) Early maneuver (telemetry on low-gain antenna);
    e) Late maneuver (telemetry on high-gain antenna);
    f) Cruise, cyclics;
    g) Retrothrust maneuver (telemetry on high-gain antenna);
    h) Solar occultation;
    i) Capsule separation sequence;
    j) Preseparation checkout sequence;
    k) Orbital operations sequence;
    l) Planetary observation sequences.

The CC&S shall be designed to conserve the number and complexity of Earth-based commands. The number of real-time Earth-based commands shall be minimized. The Flight Spacecraft and Flight Capsule shall contain on-board controllers to initiate control commands that are entered prior to the time of execution.

2.1.3.3 Guidance and Control Subsystems

The guidance and control subsystems consist of:

1) Attitude-reference subsystem;

2) Autopilot sybsystem;

3) Reaction-control subsystem.

The propulsion subsystem (thrust vector control) and the central computer and sequencer also contribute to the guidance and control subsystem.

Functional Design Restraints--The major subsection of guidance and control are attitude control and velocity control. Design characteristics and restraints for these subsections given in Table 2-1.

Table 2-1 (Attached)

Attitude Control Interface--When operating in the acquisition and cruise mode, the attitude-control subsystem shall be capable of acquiring and maintaining three-axis stabilization (using the Sun and Canopus as reference;) to a nominal accuracy of ±0.5 degree in respect to each axis. Reacquisition in the event of loss of acquisition for any noncatastrophic reason shall be automatic.

In the maneuver mode, the attitude-control subsystem, in response to commands from the guidance and control subsystem, shall be capable of pointing the propulsion-subsystem thrust axis to any commanded orientation with a 1-σ accuracy of ±0.5 degree from the nominal reference attitude and with a drift rate of less than 0.5 degree per hour. This stability will be maintained during motor burning using an autopilot with jet-vane actuation, similar to the autopilot on Ranger and Mariner spacecraft.

In the inertial reference mode, the attitude-control subsystem shall accept an inertial-position-error signal in the roll channel rather than the Canopus-sensor-position error. Roll turn-increment commands shall be given by ground command when operating in this mode.

Velocity Control--The guidance and control subsystem shall be capable of controlling the midcourse motor burn time by an accelerometer and timer.

Guidance and Control Equipment--Trade offs between timer-oriented and memory-oriented special-purpose computers shall be performed.

Memory and Logic Elements--Only semiconductor and ferrite-core logic and memory elements will be considered.

Control Devices--A passive control stabilization such as derived rate will be used in the interplanetary cruise phase.

Gyros will be used in a rate mode for rate reduction and control-system compensation in the acquisition mode. Passive compensation may be used as a backup.

1) Single-axis floated integrating ball-bearing gyros or gas-bearing gyros shall be considered. Pulse-torque rebalance loops will be considered. Spin-motor power requirements will not be less than 1.5 watts. Torquer scale factor will not exceed 400 degrees per hour per milliampere.

2) Two-axis gyros with torquer-to-rebalance capability will be considered.

Star Sensors--Star sensors will not consist of moving parts that are subject to wear or cold-vacuum welding. Sensors will use a photovoltaic, photoconductive, or photoemissive detector.

Actuators--Flex leads will be used in preference to slip rings; geared actuators will be able to withstand stalled conditions at the output shaft without internal damage; and mechanisms will be sealed and pressurized with inert gas.

Cold-Gas Reaction Jets--For maneuvers that have very low thrust levels and which involve complex duty cycles and where the system mass is a very low fraction of the total spacecraft mass, cold-gas systems are preferred.

2.1.3.4 Propulsion--Midcourse

The Planetary Vehicle's trajectory will be altered by the midcourse-propulsion subsystem. This subsystem will provide the necessary impulse to maneuver the spacecraft into the desired trajectory.

Midcourse-Propulsion Requirements--The Planetary Vehicle will have the capability to perform at least three midcourse corrections plus one backup maneuver. A sufficient midcourse velocity allocation will be provided to obtain a nominal trajectory for the spacecraft within a tolerance of ± 500 kilometers. Within 2 to 10 days following injection, the first trajectory-correction maneuver will be performed. One or more subsequent maneuvers may be required to improve the aiming-point accuracy or flight-time accuracy.

The first midcourse correction will require 15 meters per second, assuming 1-σ limit to correct for random errors arising from Centaur injection-guidance dispersions. The recommended midcourse fuel will be sufficient for a minimum of 75 meters per second.

The propulsion subsystem in a fueled and pressurized condition will be capable of withstanding an ambient temperature range of 40 to 135°F from preflight through termination of the orbit-change maneuver.

The propulsion subsystems must be capable of storage in a space environment through the time of orbit change maneuver without causing a deleterious effect upon its own performance.

Leakage from the midcourse propulsion subsystem in a vacuum environment will be minimized by isolating pressurized portions of the propulsion subsystem. The propulsion subsystem will be designed to be safe and operable for any single condition of potential malfunction within a subsystem circuit. The ignition and thrust termination signals will be provided by the central computer and sequencer.

Orbit-Insertion Propulsion--The orbit-insertion propulsion subsystem will provide a velocity increment (ΔV) of 5700 $\pm$ 20 feet per second for a 5500-pound spacecraft. The solid motor shall be positioned so that its exhaust plume will not impose intolerable thermal loads on the spacecraft appendages.

2.1.3.5 Thermal Control

The spacecraft and the thermal control Subsystem will be designed to maintain all component parts within specified temperature limits throughout the entire spacecraft history, including ground checkout, launch, and all phases of the flight sequence.

Louver assemblies will be designed to control the heat rejected to space from electronics bays, science platform, and spacecraft propulsion similar to the Orbiting Astronomical Observatory per NASA. In the nominal setting, the louvers are fully closed at 50°F and fully open at 80°F.

Thermal shields and insulation will be used to isolate spacecraft element groups from space and from one another. Solar shields will be used to reduce the effect of changing solar flux during the transit phase on the Spacecraft Bus and certain external components (i.e., magnetometer). The capsule, spacecraft internal equipment, and spacecraft propulsion will be separated by adiabatic interfaces. Heat shields will isolate the engines to prevent an influx of heat resulting from engine firing.

Surface finishes affecting heat exchange between elements and space will not be dependent on coatings and finishes that may have uncertain properties as a result of electromagnetic or corpuscular particle damage. High reflectance, low emittance surfaces will be used to achieve maximum exchange factors.

Controlled electric heat will be used where necessary to ensure adequate temperature control margins and to obtain fine temperature control especially in the case of scientific sensors during operations. Heaters will be controlled automatically by thermostatic bimetal thermal switches. Consideration is being given to heater circuits which may be actuated by a preprogrammed sequence implemented in CC&S by ground command. Use of this concept will depend on analysis and test of the final spacecraft configuration.

The Spacecraft System and its component parts will be tested in a space chamber to provide the maximum assurance, before launch, that the design objectives are satisfied.

Consistent with special equipment requirements, the electronic assemblies will be suitably mounted to provide conductive heat paths to the spacecraft. Any subassembly within an assembly will be capable of serving as a heat path or sink for heat loads generated within other subassemblies. Adjacent subassemblies will use radiative and conductive heat exchange to the maximum extent. The assembly chassis will provide a surface suitable for application of required temperature control finishes. Surface flatness and the number of fasteners used will be compatible with thermal control requirements.

2.1.3.6 Structures and Mechanisms

The equipment required to perform the Voyager flight functions will be assembled into a unified structure that will facilitate operation as a complete system. The structure will provide load and thermal paths, suitably rigid support points for interface items, and protection against environmental factors.

Ground handling and transportation loads will not control the design of the spacecraft structure, except for local handling fittings. Limit loads for the spacecraft structure shall be maximum anticipated flight loads.

Flight loads shall be determined by analysis of the complete vehicle. The following conditions and events shall be considered:

1) Internal pressure
2) Liftoff
3) External pressure reduction with altitude
4) Maximum dynamic pressure
5) Thrust termination (all stages)
6) Mars transit
7) Capsule separation
8) Mars-orbit injection-engine ignition
9) Mars-orbit injection-engine cutoff

Spacecraft separation-joint preloads will be considered separately and in combination with launch loads. Pressure-vessel loads due to internal pressure and launch shall be considered separately and in combination. Axial and lateral launch loads shall be combined as indicated by the design trajectory. The effects of any coexistent thermal environment shall be included in the load analysis where appropriate. Dynamic buckling shall be considered for members subjected to oscillatory loads.

The spacecraft shall be positively secured to the upper part of the launch vehicle adapter by a simple mechanical system that provides adequate structural continuity during the boost phase.

The center-of-gravity limitation of the Planetary Vehicle in the boost-mode configuration is a semi-infinite cylinder 3 inches in radius, with the centerline on the vehicle roll axis, and with the upper end of the cylinder at vehicle station 2170.00.

The spacecraft adapter will have an inflight mechanical disconnect system, a pull-apart electrical connector, and a prelaunch separated electrical connector, as well as electrical cabling from the two connectors above to a connector at the field joint.

The structures and mechanisms subsystem shall be designed to:

1) Support spacecraft assemblies;
2) Support spacecraft components;
3) Maintain adequate alignment between components;
4) Provide acceptable static and dynamic load environments;
5) Support flight capsule;
6) Separate the spacecraft from ground-launch facilities (umbilical, etc.);
7) Separate the spacecraft from the launch vehicle;
8) Point articulated antennas;
9) Point instruments;
10) Extend and/or erect stowed members, e.g., solar panels, protective covers, instrument supports, etc.;
11) Jettison used spacecraft parts;
12) Release capsule from spacecraft.

Structural and Mechanical Design Criteria--Details are outlined below.

Design considerations--Structural design considerations are given in functional specification MC-4-521 A, "Functional Specification," Mariner C Flight Equipment.

Structural Design Criteria--A prime design consideration is the coupling of structural modes during launch, midcourse, and orbit-insertion sequences. During launch it is necessary to consider the dynamic structural interaction of the launch vehicle and the Planetary Vehicle.

Design Loads--The design loads for a spacecraft structure development program will be based on a rational consideration of the loads applied to the composite spacecraft-launch vehicle structural system. Those loads will be upgraded by an iterative approach applied throughout the development period. The design loads of Table 2-2 are given to provide a uniform basis for preliminary structural design.

TABLE 2-2: Design Loads

Condition	Static		Vibration		
	Longitudinal	Lateral	Longitudinal	Lateral	Torsion
			0-peak	0-peak	0-peak
	g	g	g	g	rad/sec^2
(1)	6	1	0.8	0.5	0
(2)	2	1	1.2	0.75	60
(3)	1	0	1.6	1.0	60
(4)	0	0	1.6	0	0

The combined longitudinal and lateral static loads shall be combined with vibration in one or more critical directions. All lateral loads shall be considered in the most critical lateral direction.

All vibration inputs shall be considered as discrete transients which may occur at frequency for the duration shown in Table 2-3. Additional criteria appear in Section 2.1 of Boeing Document D2-82729-1, "Structural Load, Analysis, and Test Data for Voyager Phase IA Study."

Table 2-3: LENGTH OF VIBRATION TRANSIENT

Direction	Vibration Frequency Range (cps)		
	2.5-10	10-40	40-160
Axial	40 cycles	30 cycles	0.5 sec
Lateral	40 cycles	30 cycles	0.5 sec
Torsion	20 cycles	20 cycles	0.25 sec

Environmental requirements pertaining to shock and random vibration shall also be considered when applicable.

Structural Analysis and Test Requirements--The analytical model of the spacecraft structure shall be sufficiently comprehensive to approximately determine frequencies, mode shapes, deflections, and critical stresses. Separation dynamics shall also be considered. Structural test procedures necessary to supplement the analytical work to qualify the spacecraft structure shall be performed.

Mechanisms--The use of explosive devices shall be minimized. The alternative of mechanical or electromechanical devices is preferred.

Solar Panels--Rigid, flat, or hinged solar-panel structures may be considered. Wrap-around, flexible, roll-up, or structures with extension booms will not be considered. Panels shall not be designed to refold or retract in flight.

Pressure Vessels--Vessels which will be gas pressurized in the vicinity of personnel shall be fabricated of Ti-6Al-4V titanium alloy in the annealed condition. For small low-pressure vessels, 6061-T6 aluminum is acceptable. A hazard factor of 1.76 (2.2 ultimate) shall be used. Vessels which will not be pressurized in the vicinity of personnel may be fabricated of Ti-6Al-4V titanium heat-treated to 165,000 psi maximum yeild strength.

Weights--The flight spacecraft less the Flight Spacecraft adapter shall not exceed 5500 pounds, including 250 pounds of Flight Spacecraft Science per JPL Payload. Weight margins shall be carried in all weights during system design, and they shall be identified in all weight statements. Margins shall be consistent with weight estimation confidence levels. Weights shall be reported separately by functions, by subsystems, and by the equipment list.

Sufficient flexibility shall be available within the structures and mechanism subsystem design to accommodate requirements of specific payloads as ultimately selected for each Voyager mission.

Mountings--Special mounting provisions and hardware are required for items such as the magnetometer boom and antennas.

The Science Payload instrument per JPL interface requirements with the structures and mechanism subsystem will include mechanical attachment, electrical attachment, adequate fields of view, and adequate isolation (radiation, vibration, magnetic fields, etc.) from the Spacecraft Bus.

Isolation will include various covers which must be removed as part of the flight sequence. These covers will be on the Spacecraft-Bus side of the interface. In addition, the planet-oriented instruments will require support from the Spacecraft Bus consisting of planet-sensing and pointing-control functions. Additional scan modes will be accomplished within the Spacecraft Science Payload itself. The data automation equipment and some of the instrument electronics will be made up of standardized electronic modules and will be housed in one of the Spacecraft Bus equipment packages.

In accordance with JPL specifications, three surface observation instruments will be mounted on a common scan platform; individual mounting on separate platforms may be necessary for two atmospheric observation instruments. There will be six planetary-interplanetary environment observation instruments. All but one of these instruments will be affixed to the body of the Spacecraft Bus; the one exception will be mounted to achieve magnetic isolation from the body of the Spacecraft Bus to the maximum extent permitted by the spacecraft design. Antennas will be required for some experiments.

2.1.3.7 Electrical Power

Flight Spacecraft power during the launch phase and the solar acquisition phase will be provided by batteries. Subsequent to solar acquisition, power will be provided by solar panels. Power interfaces will be established between the Data Automation Equipment and the Spacecraft Bus. The Flight Spacecraft will be required to provide instantaneous peak power not to exceed 200 watts to the Flight Capsule during interplanetary cruise. Specified engineering measurements will be made during flight for the purpose of evaluating the operation of the power subsystem and to verify that certain functions are being performed.

Power Subsystem Requirements--The power subsystem shall be designed to:

1) Provide electrical power from the primary power source for spacecraft operation;
2) Provide electrical power from a charged battery when the primary source will not handle the load;
3) Provide power from the primary source to recharge the battery;
4) Condition power for spacecraft use; parameters to be regulated are voltage, frequency, wave-form, phase, and noise level;
5) Distribute power as required;
6) Condition power to individual requirements.

Primary Power--Primary power shall be provided by solar panels. In the solar panel design, values greater than those presented below shall not be used.

1) Photovoltiac solar cells shall be of the silicon single-crystal variety, with dimensions no greater than 3 x 3 cm.

2) Cell-packing factors greater than 90% shall not be considered.

3) Specific power capability of solar arrays shall not exceed 11 w/ft^2 at a panel temperature of 57°C, oriented normal to a space Sun intensity of 140 mw/cm^2.

4) Solar cells thinner than 8 mils shall not be considered.

5) Solar panel structures shall be designed for an α/E no greater than 0.5 radiating from both sides to free space.

An upper limit to performance characteristics of primary batteries shall be as follows:

1) Sealed Zinc Silver-Oxide

    95 percent capacity retention for 1 year at 70°F.

    100 watt-hours per pound maximum.

    Maximum battery shelf life of 2 years.

    Minimum volume of 0.5 cubic inch per watt-hour.

2) Remotely Activated Zinc Silver-Oxide

    100 percent capacity retention for 1 year at 70°F.

    50 watt-hours per pound maximum.

    Maximum battery shelf life of 5 years.

    Minimum volume of 0.5 cubic inch per watt-hour.

Secondary Power--An upper limit to performance characteristics of secondary batteries shall be as follows:

1) Sealed Zinc Silver-Oxide

    95 per cent capacity retention for 1 year at 70°F.

    50 watt-hours per pound maximum.

    Maximum battery shelf life of 2 years.

Minimum volume of 0.5 cubic inch per watt-hour.

Maximum cycle life of 500 cycles at 30 per cent depth of discharge.

Maximum depth of discharge of 95 per cent.

Maximum amp-hr recharge efficiency of 98 per cent.

2) Sealed Silver-Cadimum

95 per cent capacity retention for 1 year at 70°F.

30 watt-hours per pound maximum.

Maximum battery shelf of 2 years.

Minimum volume of 0.59 cubic inch per-watt-hour.

Maximum cycle life of 2000 cycles at 30 per cent depth of discharge.

Maximum depth of discharge of 95 per cent.

Maximum amp-hr recharge efficiency of 98 per cent.

Regulators and Inverters--An upper limit to performance characteristics of electrical regulators shall be as follows:

1) Booster Regulators

Power handling capability of 400 watts per unit, maximum.

Voltage stability of ±1 per cent.

Efficiency at 400-watt power level: 86 percent at 40 volts or 88 per cent at 25 volts.

2) Pulse-Width Modulated Series Switching Regulators

Power handling capability of 400 watts per unit, maximum.

Voltage stability of ±1 per cent.

Efficiency at 400 watt power level: 90 percent at 50 volts 94 per cent at 35 volts.

3) Shunt Regulators

Power handling capability of 100 watts per unit maximum.

Voltage stability of ±1 per cent for the extreme panel temperature variation.

4) Regulating Inverters

Power handling capability of 400 watts per unit maximum.

Voltage stability of ±1 per cent, square wave.

Efficiency of 85 per cent maximum.

An upper limit to performance characteristics of inverters shall be as follows:

1) Free running frequency accuracy of ±1 percent.
2) Frequency accuracy when stabilized of ±0.01 per cent.
3) Efficiency of 40-v input and a load power of 300 w: 90 per cent maximum.

Synchronization shall be accomplished by means of either phase lock or frequency lock. The synchronization method shall be sensitive enough to provide operation with 50 per cent distorted synchronization signal maintaining required outputs.

2.1.3.8 Pyrotechnics

Should mechanism design warrant the use of explosive devices; the pyronetic equipment shall conform with the following:

General--General Range Safety Plan--Electroexplosive devices (EED), associated system wiring, and firing circuity shall conform to AFETR 80-2, "General Range Safety Plan," Volume I, and associated Appendix A.

Standardization--Consideration shall be given to a standard squib envelope and matchhead configuration for all EED operations.

Electroexplosive Devices--The following shall apply:

1) All squibs shall contain redundant (two) bridge wires.
2) EED's shall utilize connector-type squibs, not pigtails.
3) Squib bodies shall be made of one-piece construction (the connector receptacle must be an integral part of the squib body), and shall not be vented.
4) All EED's shall utilize 1-amp/1-watt no-fire squibs (1-amp/1-watt applied to each of the two bridge circuits simultaneously).
5) Squibs shall be designed to provide continuous circumferential shielding between cable and device to ensure that the shield circuit is completed before contact is made with the bridge pins.
6) Exploding bridge wire devices shall not be used.
7) Devices will be nondetonating. Materials such as RDX shall be avoided.
8) All squibs shall be able to withstand static discharges of 25 kilovolt from a 500-picofared capacitor applied between pins, or between pins and case, at all pressures.

Initiation Circuitry--The following shall apply:

1) Redundant firing circuitry shall be used.
2) Solid-state devices shall be used for switching. No electromechanical relays shall be used.
3) The number of electrical componts shall be minimized, consistent with maximizing complete system reliability.

4) Spacecraft A.C. power shall be used for squib initiation. Direct current power system transients resulting from initiation will not produce significent effects on other equipment.

5) The design shall consider as probable the instantaneous and permanent electrical shorting of each and every squib upon firing.

2.1.3.9 Spacecraft Science Subsystem

The Spacecraft science instruments objectives will include planetary observations and planetary-interplanetary environmental observations.

The Spacecraft science hardware will consist of five planetary observation instruments: two for planetary atmospheric observations and three for surface observations. In addition, there will be six planetary-interplanetary environment observation instruments.

The scientific instruments with fixed view angles shall be located on the Spacecraft Bus. Scientific instruments that require pointing will be on a scan platform. Sensors requiring isolation from the spacecraft will be mounted on a deployable boom.

TV and Mars scanner experiments should be capable of repeating measurements on certain areas for possible detection of seasonal changes after periods of 1 to 3 months.

Instruments shall be capable of operating in the environment imposed by the spacecraft attitude-control system in the normal stabilized mode.

Instruments shall be capable of operating over the temperature range attainable by spacecraft temperature control.

All scientific instruments shall be designed to be as functionally independent of one another as is practical. Failure isolation shall be provided in all science instruments.

Data from the Science Payload will be presented to the spacecraft in a format compatible with the spacecraft data handling equipment.

The Data Automation Equipment will be designed for inclusion in the electronic packages of the Spacecraft Bus.

Calibration sequences are initiated by means of periodic commands from the Spacecraft Bus CC&S or through real-time commands from ground-based transmitters.

2.2 DESIGN RESTRAINTS

This section defines the restraints imposed on each subsystem by the system and by the other subsystems. It also includes environmental data.

2.2.1 General

All design concepts, materials, and components considered for the Voyager 1971 mission shall have a development freeze date of July 1966. Only those design concepts that have demonstrated feasibility and will have been developed by that date shall be considered for inclusion in the Voyager 1971 mission.

2.2.2 Radiation Sources

The flux produced by artificial or natural radioactive material on board the spacecraft must not produce an increase in the counting rate of the interplanetary radiation instruments in excess of 1.0 percent of the interplanetary background rate.

2.2.3 Magnetic Interference

The total magnetic field from d.c. through 5.0 cps of an assembly or subassembly shall not exceed 5 gammas measured at 3 feet. Magnetic fields above a frequency of 5.0 cps are of minor significance. The current loop contribution to the spacecraft magnetic field shall at no time exceed 0.5 gamma, measured at 6 feet from the spacecraft centerline.

The total magnetic field of an assembly or subassembly shall not change by more than 0.5 gamma, measured at 3 feet, under the following conditions:

1) As a result of Type Approval vibration testing;
2) When change between Mode 2 and Mode 3 operation occur (power off to power on)
3) During operation in Modes 2 or 3.

Magnetic Fields--Use of all materials, processes, and parts shall be controlled in such a manner that the summation of all magnetic fields shall not exceed 1.0 gamma, measured at the magnetometer in its deployed location, under simulated functional environment.

To the extent possible, according to priority with competing characteristics, materials shall be those selected by Boeing and approved by JPL which, when subjected to suitable test, exhibit permeability no greater than 1.004.

A number of materials exhibit subtle variations in magnetic behavior that must be considered in the design, fabrication, and usage of Voyager hardware. For example:

1) Stainless steels such as 304 may be paramagnetic when they are in the fully annealed condition, but become ferromagnetic if work hardened. The degree of ferromagnetism is a function of the amount of cold work, and the total resulting magnetic field is a function of the volume of metal which is cold worked.
2) Nickel-base superalloys may be paramagnetic in some heat-treat conditions and ferromagnetic in others. The heat-treat condition

for most desirable magnetic properties may not satisfy the requirements for strength, hardness, fabrication, or other considerations. Available data indicate that there is variation in magnetic properties between different samples in lots of the same material in purportedly the same condition.

The requirement for maximum magnetic cleanliness can be met providing that magnetically clean materials can be prevented from becoming magnetically contaminated in processing and in various other stages of use. This will require close attention to all phases of processing.

The environment during manufacture, mixing, and blending of raw materials must be controlled to prevent inclusion of ferromagnetic materials. Similar care must be exercised during fabrication of hardware to prevent inclusion, occlusion, or imbedment of ferromagnetic material. For example, ferromagnetic chips or filings could seriously affect otherwise magnetically clean aluminum parts. The summation of magnetic permanence is considered to be the value to be minimized.

<u>Deperming and Mapping</u>--Associated with both deperming and mapping operations, fields larger than the Earth's field will be encountered. Normal perming and deperming procedures will involve 60-cycle AC magnetic fields as high as 100 gauss (oersteds). In designing spacecraft circuitry, the requirements for subsequent deperming shall be given consideration and the enclosed area of circuit loops held to a minimum.

Umbilical and Direct Access Functions--Adequate isolation shall be provided on all umbilical and direct access functions as a safeguard against external faults.

2.2.4 Environment

The environmental constraints include the scope of manufacturing and shipping, test and launch, cruise, orbit, and Mars surface areas. An entry of NA in the following environmental tables indicates a non-applicable estimate; an entry of an X indicates an estimate of an environment is included in the text; a blank represents an environment for which there is no estimate currently available.

2.2.4.1 Manufacture and Shipment Environment

Table 2.2-1 shows the environmental considerations in manufacture and shipment.

Table 2.2-1: ENVIRONMENTAL CONSIDERATIONS IN MANUFACTURE AND SHIPMENT

	Manufacture	Bench Test	Potting & Conformal Coat	Repair	Shipping, Hand Carry	Shipping, Commercial	Shipping, System	Storage
Temperature					X	X	X	X
Humidity						X	X	X
Vibration	NA	NA	NA	NA				NA
Shock	NA	NA	NA	NA				NA
Pressure								NA
Contaminants		NA	NA	NA				
Solvents & Chemicals		NA			NA	NA	NA	NA
Magnetics	X	X	X	X	X	X	X	X

Temperature--Temperature of components or assemblies after manufacture is completed will not exceed flight acceptance testing limits. Temperature of the spacecraft during shipment and storage will be maintained between 35° and 100°F.

Silver-cadmium batteries will be stored between 20° and 50°F at the ETR prior to installation in the spacecraft.

Humidity--The spacecraft environment relative humidity will not exceed 50 percent during shipment and storage.

2.2.4.2 Test and Prelaunch Environment

Table 2.2-2 establishes the environmental conditions during test and prelaunch.

Table 2.2-2: ENVIRONMENTAL CONSIDERATIONS IN TEST AND PRELAUNCH OPERATIONS

	Subsystem Labs	SAF	ESA	On-Pad
Humidity		X		
Temperature				
Vibration	NA	NA	NA	
Shock				NA
Electrical Transients	NA	X	X	X
Corrosive Atmosphere	NA	NA		
Contamination				
ETO	NA	X	X	X
EMI	X	X	X	X
Deperming and Mapping	NA	X	X	NA
Explosive Atmosphere	NA	NA	NA	

Temperature of the spacecraft will be maintained between 35°F and 100°F prior to installing the batteries. After installation, the spacecraft will be maintained between 35 and 75°F. Temperature of the propellants will be controlled to 65°F $\pm$ 2°F during loading in the spacecraft; this tolerance is required for close control of ullage volume and suppression pressure.

Humidity--The spacecraft environment relative humidity will not exceed 50 percent.

Electrical Transients--Electrical transients can be expected as an environment on power, signal, and control cables.

ETO; SAF Through On-Pad--For microbiological decontamination, an environment of ethylene oxide-Freon 12 that conforms to JPL specification GMO-50198-ETS will be used.

Electromagnetic Radiation Environment--The rf power density levels for spacecraft test areas may be high enough to provide interference to a spacecraft or its checkout. Although rf environment will vary depending upon the facility used, spacecraft design is planned to tolerate adverse rf interference to a maximum practicable extent.

2.2.4.3 Launch Environment

Table 2.2-3 establishes the environmental considerations during launch.

Table 2.2-3: ENVIRONMENTAL CONSIDERATIONS--LAUNCH

	Engine Ignition to L + 10 Sec	L + 10 Sec to Transonic	Transonic	Transonic to S-IB Staging	Shroud Separation	S-IVB Operation	Centaur First Burn	Parking Orbit	Centaur Second Burn	Spacecraft Separation
Shock		NA	NA		X	X	X	NA	X	X
Vibration, low frequency	X	X	X	X	NA	X	X	NA	X	NA
Vibration, random	X	X	X	X	NA	X	X	NA	X	NA
Acoustic	X	X	X	X	NA	NA	NA	NA	NA	NA
Static accel-eration	X	X	X	X	NA			NA		NA
Temperature & Ther-mal transients	X	X	X	X	X	X	X	X	X	X
Pressure reduction	NA	X	X	X	X	NA	NA	NA	NA	NA
Electromagnetic Interference	X	X	X	X	X	X	X	X	X	X
Electrical transients	X	X	X	X	X	X	X	X	X	X
Electrically con-ductive gas	NA	NA	X	X	NA	NA	NA	NA	NA	NA
Electrostatic charge & discharge	NA	X	X	X	X	NA	NA	NA	NA	NA
Contamination, particulate	X	X	X	X	X	NA	NA	NA	NA	NA
Solar radiation	NA	NA	NA	NA	X	X	X	X	X	X
Albedo	NA	NA	NA	NA						

2.2.4.4 Postlaunch

Table 2.2-4 establishes the environmental considerations during postlaunch.

Table 2.2-4: ENVIRONMENTAL CONSIDERATIONS--POSTLAUNCH

	Midcourse Maneuvers	Orbit Insertion	Orbit Trim	Bus Deflection	Sterilization Canister Opening	Capsule S/C Separation
Vibration						
Shock						
Thermal Transients						
Charge Buildup (from combustion)	X	X	X	X	NA	NA
Exhaust Gases	X	X	X	X	NA	NA
Microbial Contamination	NA	NA	NA	X	X	X

<u>Charge Buildup from Combustion</u>--Potential voltages may build up on the spacecraft from the burning of the trajectory correction motor. Because there does not appear to be a discharge possibility in flight, this accumulation of charge may be quite large and may be hazardous during the cruise phase. Further study of this pnenomenon is currently under way.

<u>Exhaust Gases</u>--There may exist the possibility of gases enveloping the spacecraft during the propulsion motor burn. The actual conductivity value will have to be determined at a later date.

2.2.4.5 Environmental Characteristics

Pressure Reduction During Ascent*--The assumed nominal environment is given in Table 2.2-5

Table 2.2-5: AMBIENT PRESSURE VERSUS TIME

Time of Flight (sec)	Ambient Pressure (psia)
0	14.5
10	14.0
20	13.2
30	11.8
40	9.5
50	6.6
60	4.3
70	2.5
80	1.4
90	0.7
100	0.4

The spacecraft shall be capable of withstanding the effects of boost depressurization associated with the pressure schedule in the above table.

* Ref.: "Mariner Mars 1964 OTN Metal Shroud Systems Analysis," Report 1 MSC-A 652810, November 23, 1964.

Low-Frequency Vibration--Estimated flight vibration shall be the following sine wave:

Lateral	0.6-g rms	5 to 200 cps
Axial	1.2-g rms	5 to 200 cps

Random Vibration*--Liftoff and Transonic--The vibration environment, with the exception of low frequency, shall be assumed to be the following omnidirectional input to the spacecraft separation plane:

1) PSD peaks of 0.07 g^2/cps ranging from 100 to 1500 cps with a 6 db/octave rolloff in the envelope defining peaks below and above these frequencies.
2) Maximum total time is 60 seconds.

Acoustic Sound Field*--Liftoff and Transonic--The maximum acoustic field, for either liftoff or transonic, shall be assumed to be a reverberant field as follows:

1) Overall SPL is approximately 142 db (re $2.10^{-5} N/m^2$);
2) SPL of 133.5 db/third octave from 85 to 250 cps;
3) Rolloff at 11 db/octave below 85 cps;
4) Rolloff at 5 db/octave above 250 cps;
5) Total duration is about 2 minutes.

Static Acceleration*--The following shall apply:

1) Liftoff Through Saturn IB Staging--The longitudinal acceleration build-up to a maximum value of approximately 4 g's at first-stage

*Ref.: "Mariner Mars 1964 OTN Metal Shroud Systems Analysis," Report 1 MSC-A 652810, November 23, 1964.

shutdown. The lateral acceleration during this same period of time shall have a maximum value of 2<g's.

2) Saturn IVB Operation--The longitudinal acceleration will build up to a maximum value of approximately 2.6 g's at the end of S-IV operation. The lateral acceleration during this same period of time will have a maximum value of < 2 g's.

3) Centaur First Burn--The longitudinal acceleration will build up to a maximum value of approximately 1.0 at the end of Centaur first burn. The lateral acceleration during this same period of time will have a maximum value of < 2 g's.

4) Centaur Second Burn--The longitudinal acceleration will build up to a maximum value of approximately 2.2 g's at the end of Centaur second burn. The lateral acceleration during this same period of time will have a maximum value of < 2 g's.

<u>Temperature and Thermal Transients</u>*--From engine ignition through shroud separation, the maximum heat rate from the shroud to spacecraft shall be assumed to be 40 watts per square foot.

For S-IVB operation through spacecraft separation, the maximum aerodynamic heat rate shall be assumed to be 24.2 watts per square foot.

2.2.4.6 Planetary Upper Atmospheres and Interplanetary Space

<u>Near-Earth</u>--The Earth atmosphere described by the U. S. Standard Atmosphere, 1962, shall be used as reference. At high altitudes, variations

*Reference: "Mariner Mars 1964 OTN Metal Shroud Systems Analysis," Report 1 MSC-A 652810, November 23, 1964.

in the observed density from the model atmosphere, which are as large as a factor of 5, may occur because of variation of solar activity and because of diurnal and seasonal variations. Similar variation in the pressure may result.

Cruise--The number density of interplanetary matter is approximately 100 particles per cubic centimeter. This matter is composed primarily of hydrogen, hydrogen ions, helium, and helium ions. The density varies with solar activity and, in addition, probably decreases somewhat with increasing distance from the Sun.

Flyby--The Martian upper atmosphere, especially the exosphere, is not well defined at this time. Until more definitive scientific measurements and interpretations can be made, the following atmosphere maximum density estimate is provided. Circular and elliptical orbit are the same as fly-by.

Up to 1500 kilometers from the Martian surface, the atmospheric parameters are assumed to have values given by Model 1 of NASA engineering models of Mars atmosphere for entry vehicle design. The atmospheric density above this level is shown in Table 2-9.

2.2.4.7 Radiation--Solar Thermal Radiation

Near-Earth--The solar spectrum, outside the Earth's atmosphere, shall be assumed to have the shape of the Johnson curve (Johnson, F. S., "The Solar Constant," Journal of Meteorology, Vol. II, No. 6, pp 431-439.)

Table 2.2-6: ATMOSPHERIC DENSITY

Altitude (km)	Density (gm/cm^3)
1,500	2.7×10^{-4}
2,000	1.5×10^{-14}
3,000	6.9×10^{-15}
4,000	4.0×10^{-15}
5,000	2.5×10^{-15}
6,000	1.7×10^{-15}
8,000	1.0×10^{-15}
10,000	7.2×10^{-16}

and an integrated intensity of 127 watts per square foot at a distance of one astronomical unit.

Cruise--During cruise, solar radiation will continuously vary from that of near-Earth to that near-Mars.

Near-Mars at Encounter--At Mars, the solar spectrum shall be assumed to have the shape of the Johnson curve, and will vary in intensity from 57.2 to 67.0 watts per square foot, depending upon the Sun-Mars distance, which varies somewhat during the 1971 encounter period.

Mars Orbiter--During the 6 month following encounter, the Sun-Mars distance changes. Therefore, the solar spectrum is as given for near-Mars with an extreme spread of integrated intensities from 67.0 watts/sq.ft. at encounter to 46.7 watts/sq.ft. 6 months later for 1971.

2.2.4.7.1 Corpuscular Radiation

Geomagnetically Trapped Particle Radiation--This radiation consists of the following:

1) Near-Earth--Omnidirectional flux along the geomagnetic equator:

   Peak proton flux ($E > 30$ Mev) $= 1.5 \times 10^4 - 5 \times 10^4$

   Inner-belt ($1.5 \leq L$ 1.7) = protons/cm^2/sec

   Time-integrated proton flux $= 1 \times 10^7$ protons/cm^2 ($E > 30$ Mev)

   Peak electron flux ($E > 0.5$ Mev)

      Inner-belt ($1.3 \leq L \leq 1.4$)$= 4 \times 10^8$ electronics/cm^2/sec

      Outer-belt ($4 \leq L \leq 4.5$)$= 2.5 \times 10^6 - 1 \times 10^7$ electrons/cm^2/sec

Time-integrated electron flux = 4×10^{10} electronics/cm^2/sec
(E >0.5 Mev)

Spectral distribution - Spectral distributions of omnidirectional flux along the magnetic equator are given in Figures 2-1 through 2-3.

Angular distribution - Isotropic

a) Protons (inner belt)--variable within a factor of 2 to 10

b) Electrons (inner belt)--variable within a factor of 2 to 10

c) Electrons (outer belt)--variable within a factor of 10 to 100

2) Flyby--The design environment for magnetically trapped particle radiation around Mars will be assumed to be equivalent to the near-Earth environment. Estimated peak flux positions (measured from center of Mars) in the Martian radiation belts are 5000 kilometers for the inner belt and 16,000 kilometers for the outer belt.

3) Circular orbit--same as flyby

4) Elliptical orbit--same as flyby

The selection of near-Earth equivalent as a design environment for magnetically trapped radiation about Mars was based on a qualitative analysis considering sources of radiation, Mars magnetic field strength, Mars atmospheric characteristics, and radiation-loss mechanisms. Opinions from Dr. J. Van Allen, Iowa State University; Dr. G. de Vaucouleurs, University of Texas; Dr. Hess and Dr. S. Hennes, Goddard Space Flight Center; and Dr. S. Neddemyer, University of Washington were solicited. In every case these authorities noted that the magnetically trapped radiation at Mars can be expected to be less than that of Earth.

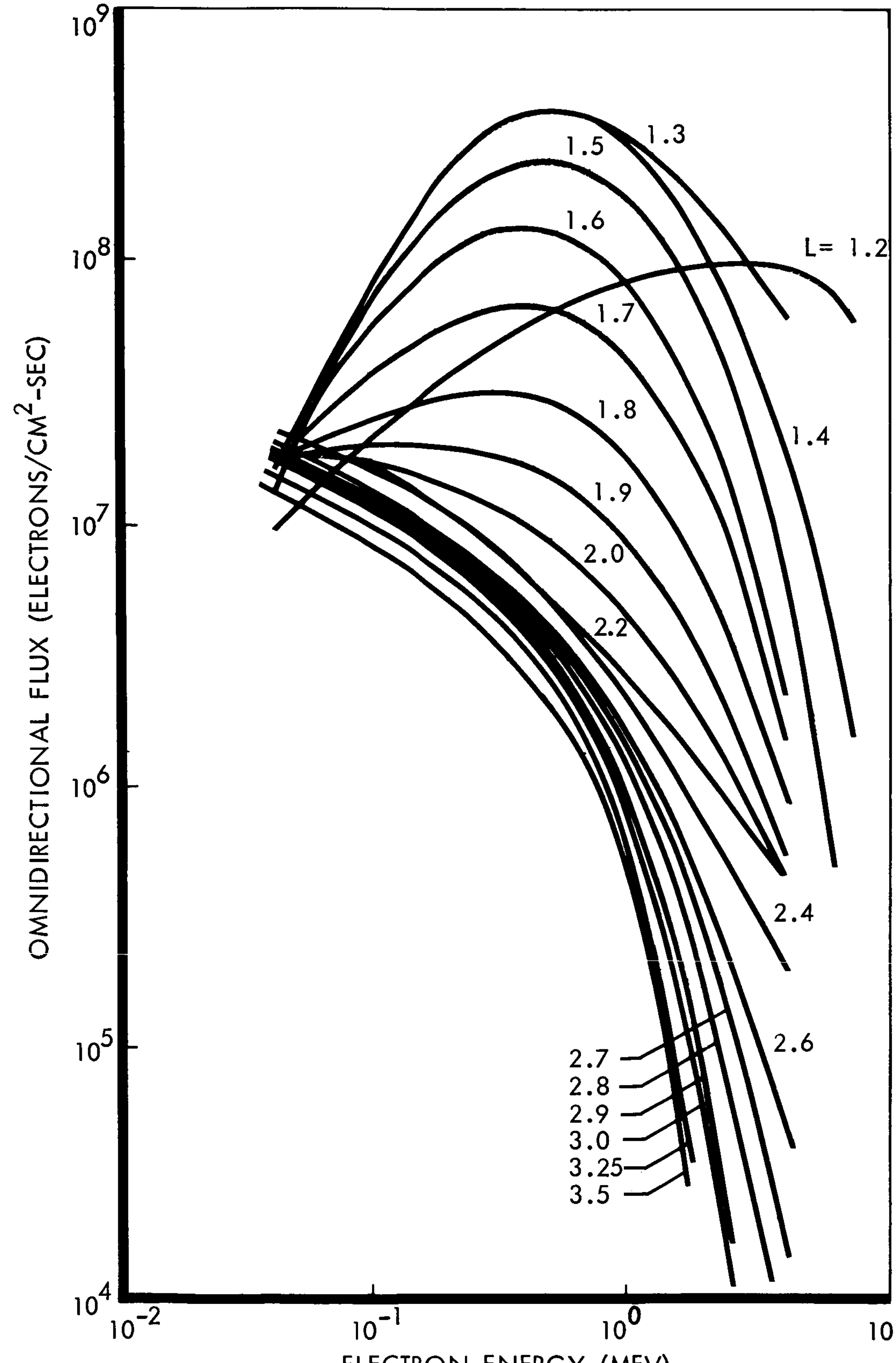


Figure 2-1: Omnidirectional Flux of Electrons Along Magnetic Equator for Various Earth Radii (L)

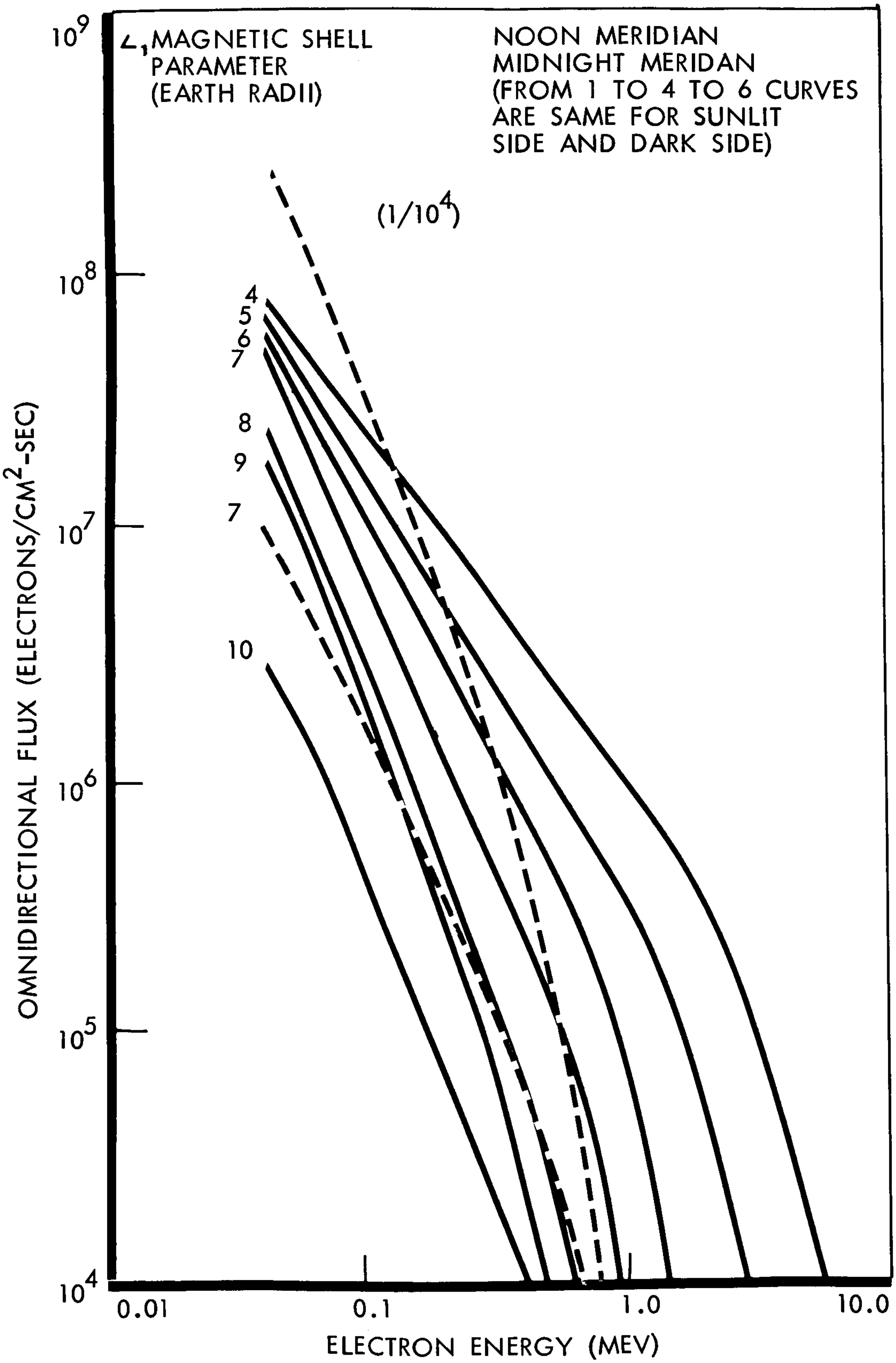


Figure 2-2: Omnidirectional Flux of Electrons Along Geomagnetic Equator for Various Earth Radii (L)

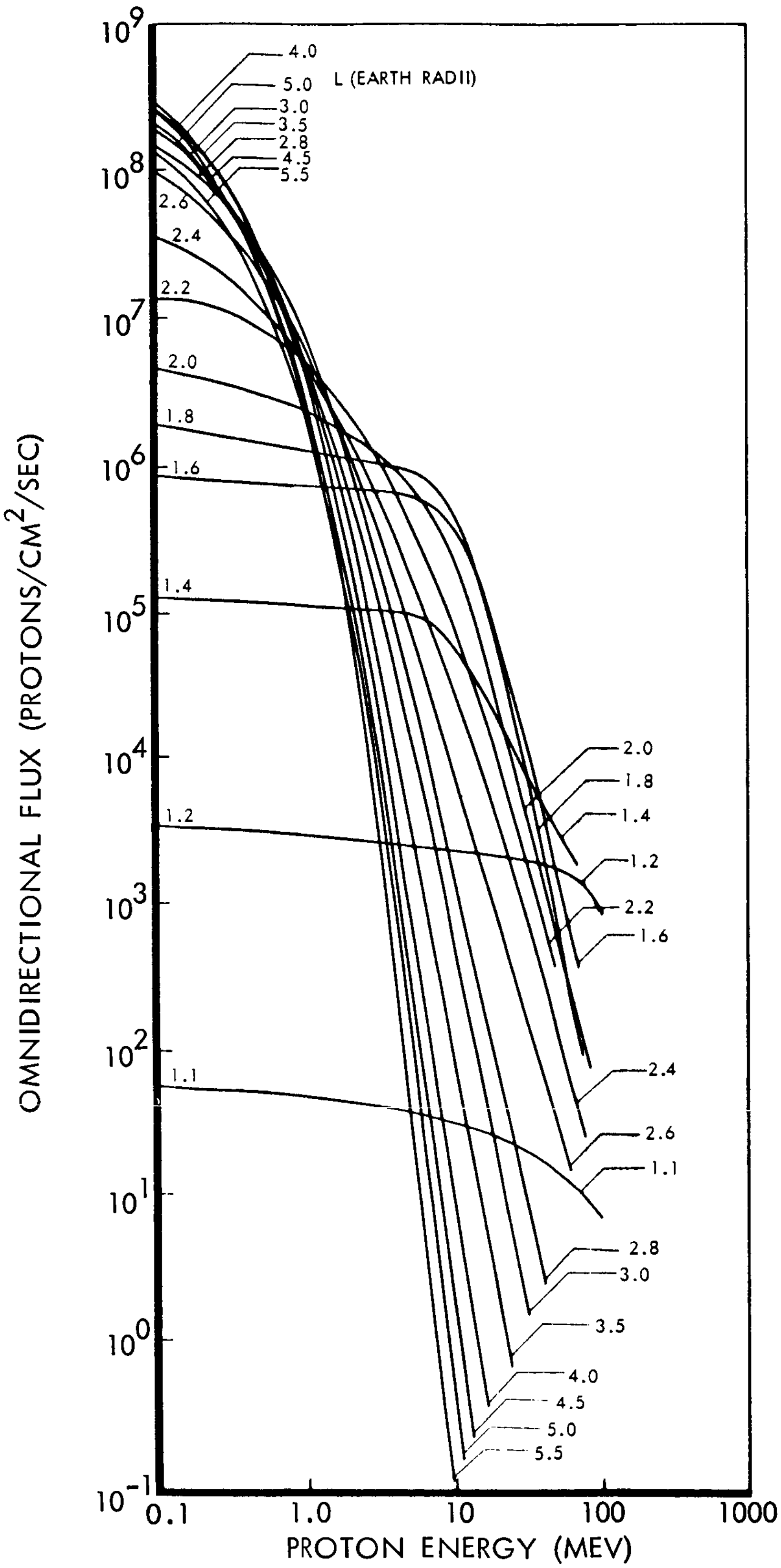


Figure 2-3: Omdidirectional Flux of Protons Along Magnetic Equator for Various Earth Radii (L)

The rationale for establishing the intensity and spectral distribution for the Mars radiation belt is based on Earth analogy. The primary component of trapped radiation in the Earth's inner Van Allen belt consists of high-energy (10 mev $<$ E $<$100 mev) protons formed from albedo neutron decay. The sources of the neutrons are nuclear disintegrations of atmospheric nuclei caused by galactic cosmic radiation. Much less is known about the outer belt source and loss mechanisms for trapped radiation. A few generalizations can, however, be made. It is clear from observations of Injun III and other satellites that the solar wind is the basic source of the particles. These electrons are injected in the magnetosphere at the magnetopause and are subsequently accelerated to auroral energies and beyond by magnetic perturbations (magnetic pumping). As particles move onward, magnetic pumping increases their energy. Large scale losses of particles can be caused by large magnetic perturbations, as verified by various satellite measurements.

The galactic cosmic radiation will be approximately the same near Earth and Mars, and the difference in composition of the two atmospheres is relatively unimportant from the point of view of nuclear collisions. Hence, the source mechanism for inner belt protons on Mars will be essentially the same as that for Earth.

Since Mars is farther than the Earth from the Sun, the solar wind pressure and density will be smaller. Furthermore, Mars has a smaller magnetic field-- less than half that of Earth. Due to the weaker solar wind, fewer particles are available at the boundary of the magnetopause, and a smaller probability of injecting these particles into the magnetosphere exists. Once these particles are in the magnetosphere, the weaker solar wind also decreases the probability of favorable circumstances necessary to energize the particles up to auroral energies and beyond. Hence, the source mechanisms for outer belt electrons are less favorable for injection around Mars than around Earth.

Relative to the inner belt protons, two major aspects of the loss processes should be considered: (1) the exospheric density, and (2) the magnitude of the magnetic flux density. High-energy protons ($E_p > 300$ Mev) are degraded in energy primarily by nuclear collisions. They lose energy and eventually are lost from the belts by continued inelastic collisions with the background exospheric gas. The lower edge of the inner belt will be truncated at an altitude where the background gas density is high and the loss rate is high. For all practical purposes, this occurs in the terrestrial trapped radiation at a nominal altitude of 500 kilometers. The outer boundary of the inner belt occurs where the magnetic flux density becomes too small to trap the high energy protons. In this case, the particles experience during one gyration a considerable change in magnetic field, and undergoes an irreversible transfer of energy to the magnetic field. Consequently, it is not trapped.

Since the specified Mars surface magnetic field is no more than half that of the Earth, the adiabatic loss rate ($\Delta B/B$) is larger on Mars than on Earth for the same scale of distances. Of much more importance is the relationship of the exospheric densities. Although the surface pressure and density on Mars are very small in comparison to those on Earth, the scale height is much larger. As a consequence, the exospheric density on Mars is much greater than on Earth. As an example, consider that the peak flux intensity for the inner belt occurs at 5000 kilometers from the center of Mars. This distance corresponds to an altitude of approximately 1500 kilometers above the martian surface at a density of approximately 2.7 x 10^{-14} g/cm^{-3}. The Earth's atmosphere has a similar density at an altitude below 400 kilometers. Hence, the heart of the inner belt on Mars lies in a region of atmospheric density greater than that of the Earth. Consequently, the inner-belt proton component on Mars must be vanishingly small, at least several orders of magnitude below the comparable terrestrial flux. Inclusion of polar-cap splash albedo and other refinements to the general knowledge of the inner belt, does not appreciably modify these conclusions.

Since so little is known about the loss mechanisms for outer-belt electrons on Earth, it is difficult to compare them to those on Mars. It is, however, still quite reasonable to assume that the loss mechanism should be greater on Mars. The weaker magnetic field should offer less possibility of containment, and the higher background gas density should offer a greater rate of energy transfer. Hence, the loss mechanisms for outer-belt electrons should be greater for Mars than for Earth.

In the case of the outer belt, the source is weaker and the loss is greater for Mars than for Earth. It is difficult to estimate how much smaller the outer belt flux will be for Mars than for the Earth, but it certainly cannot exceed the terrestrial trapped radiation. In the case of the inner belt, the source is approximately the same, but the loss rates are several order of magnitude greater on Mars than on Earth. Hence, the inner belt proton population should be quite small. Therefore, selection of near-Earth equivalence as the Martian design environment for trapped radiation belts is reasonable and very probably quite conservative.

Galactic Cosmic Radiation--This consists of the following:

1) Near-Earth

a) Primary Radiation

(1) Omnidirectional Flux

(2) Heavy particles 1.5 particles/(cm^2 - sec) during solar maximum
4.0 particles/(cm^2 - sec) during solar minimum

(3) Electrons 10^{-2} to 10^{-1} electrons/(cm^2 - sec) (E >100 Mev)

(4) Gammas 10 to 40 protons/(cm^2 - sec) (E ~ 100 - 200 kev)

Average Yearly Flux -- 7.8×10^7 particles/cm^2

Estimated Maximum Peak Yearly Flux During Year of Solar Minimum -- 1.2×10^8 particles/cm^2

Integrated Dosage -- 6 to 20 rad/year

Composition -- (see Table 2.2-7).

Table 2.2-7: ELEMENTAL COMPOSITION OF PRIMARY GALACTIC COSMIC RAYS

Element	Atomic No., Z	Percent of Total
Hydrogen	1	80 to 85
Helium	2	11 to 16
Light Nuclei (L)	$3 \leq Z \leq 5$	2
Medium Nuclei (M)	$6 < Z \leq 9$	1
Heavy Nuclei (H)	$Z \geq 10$	3

Spectral Distributions--Spectral distributions of integral flux are given in Figure 2-4 for the total primary galactic cosmic radiation and in Figure 2-5 for the various heavy particle component.

b) Secondary radiation in upper atmosphere:

(1) Omnidirectional flux

(2) Peak total flux $\sim$ 7 particles/(cm^2 - sec)

(3) Peak electrons $\sim$ 3 particles/(cm^2 - sec)

(4) Peak gammas $\sim$ 2.5 particles/(cm^2 - sec)

(5) Peak neutrons and protons $\sim$ 3 particles/(cm^2 - sec)

(6) Peak total mesons $\sim$ 0.7 particle/(cm^2 - sec)

2) Cruise:

a) Primary Radiation--As for the Earth, using solar minimum flux values and upper-limit dosages.

b) Secondary Radiation--No secondary galactic cosmic radiation.

3) Flyby--Same as cruise.

4) Circular Orbit--Same as cruise.

5) Elliptical Orbit--Same as cruise.

Radioisotope Thermoelectric Generator Radiation--This consists of the following:

1) Near-Earth--Dose rates from unshielded source (right circular cylinder, L/D = 1). Total radiation dose rates (millirad/hr) from a 2000-thermal-watt plutonium power source (estimated maximum electrical power $\sim$ 100 watts) are shown in Table 2.2-8.

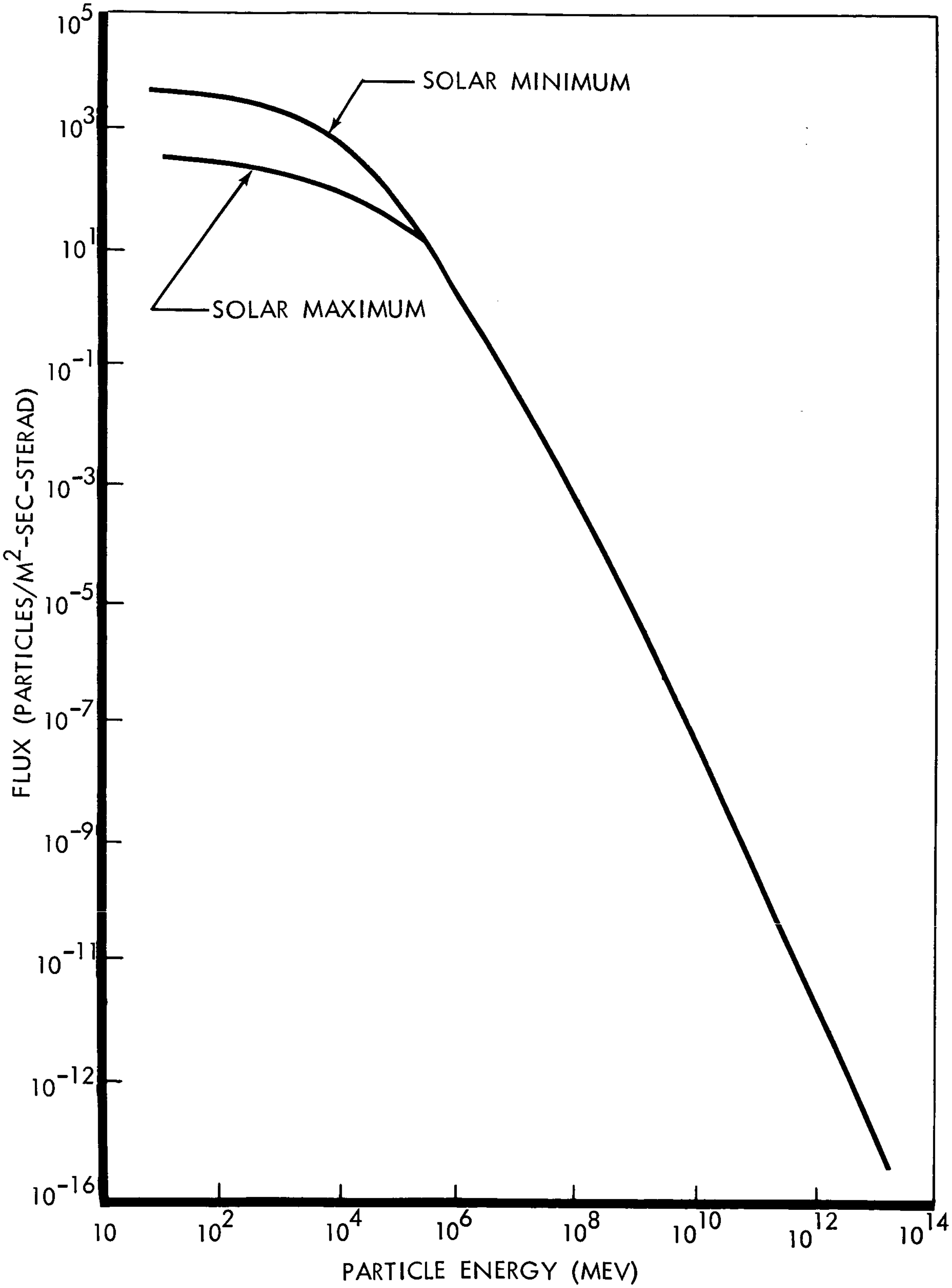


Figure 2-4: Integral Spectra for Cosmic Radiation

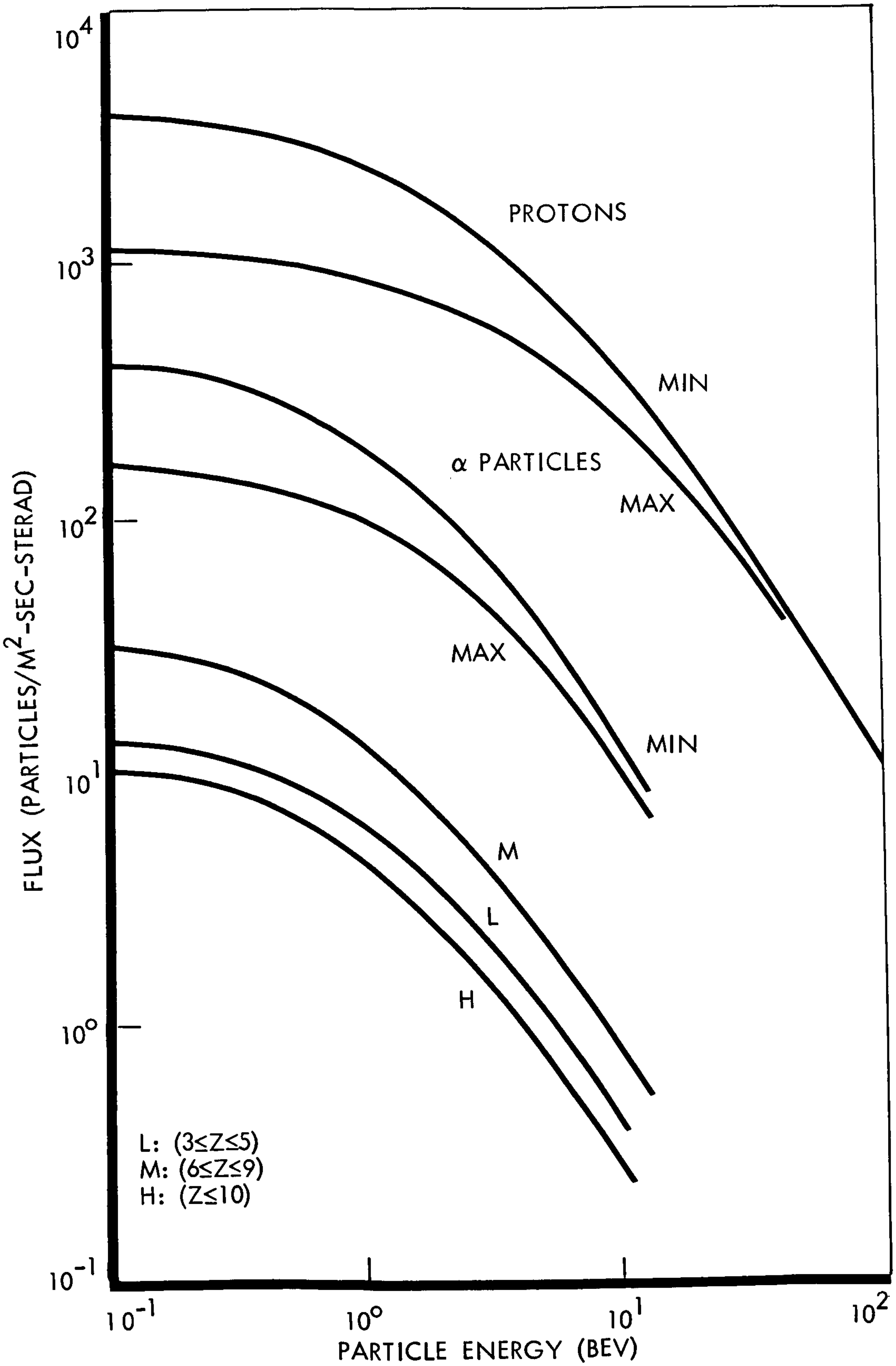


Figure 2-5: Integral Spectra for Galactic Cosmic Radiation

Table 2.2-8: RADIATION

Radiation Type (millirad/hr)	Distance from Center of Source 1 Meter	2 Meters	3 Meters
Neutrons	15	3.8	1.7
Gammas	2.9	0.7	0.3

2) Cruise--Same as for near-Earth.

3) Flyby--Same as for near-Earth.

4) Circular Orbit--Same as for near-Earth.

5) Elliptical Orbit--Same as for near-Earth.

Solar Cosmic Radiation--This consists of the following:

1) Solar Flares

a) Near-Earth

(1) Time-integrated flux is shown in Table 2.2-9.

Table 2.2-9: SOLAR FLARE EVENT ENVIRONMENTAL REQUIREMENTS

Parameter	Requirement
Peak proton flux ($E > 30$ Mev)	1.3×10^4 protons/(cm^2 - sec)
Peak proton flux ($E > 100$ Mev)	3.8×10^3 protons/(cm^2 - sec)
Time-integrated proton flux per flare ($E > 30$ Mev)	1.0×10^9 protons/cm^2
Time-integrated proton flux per flare ($E > 100$ Mev)	2.6×10^8 protons/cm^2
Time-integrated proton flux per year ($E > 30$ Mev)	1.0×10^{10} protons/(cm^2 - yr)
Time-integrated proton flux per year ($E > 100$ Mev)	9.6×10^8 protons/(cm^2 - yr)

(2) Spectral Distribution--Power-law representation using above data at 30 and 100 Mev.

b) Cruise--As for near-Earth, using $(1/R)^2$ of distance in A.U. from Sun for implied spatial continuity.

c) Flyby--Same as for cruise.

d) Circular orbit--Same as for cruise.

e) Elliptical orbit--Same as for cruise.

2) Solar Wind

a) Near-Earth

(1) Mean values of solar wind

(a) Mean Density	0.5 A.U.	20 hydrogen atoms/cc
	1.0 A.U.	five hydrogen atoms/cc
	1.75 A.U.	two hydrogen atoms/cc
(b) Mean Flux	0.5 A.U.	$8x10^8$ hydrogen atoms/(cm^2 - sec)
	1.0 A.U.	$2x10^8$ hydrogen atoms/(cm^2 - sec)
	1.75 A.U.	10^8 hydrogen atoms/(cm^2 - sec)

(2) Mean velocity of solar wind from 0.5 A.U. to 1.75 A.U. = 450 to 500 km/sec.

(3) Electron Flux: 10^3 electrons/Cm^3 for energies of a few electron volts.

b) Cruise--Same as for near-Earth for the proton components.

c) Flyby--Same as for cruise.

d) Circular Orbit--Same as for cruise.

e) Elliptical Orbit--Same as for cruise.

Auroral Radiation--This consists of the following:

1) Near-Earth

a) Peak proton flux ($E > 100$ kev)--10^6 protons/(cm^2 - sec)

b) Electron Flux

(1) Active displays--10^{11} to 10^{12} electrons/(cm^2 - sec) with energies near ~ 6 kev.

(2) Quiescent displays--10^{10} electrons/(cm^2 - sec) between 30 ev and 1 kev.

2) Cruise--No auroral radiation.

3) Flyby--No auroral radiation.

4) Circular orbit--No auroral radiation.

5) Elliptical orbit--No auroral radiation.

2.2.4.7.2 Charged-Particle Radiation

Figures 2-6 and 2-7 show the radiation doses, dose rates, particle fluxes, and fluences for which the spacecraft will be designed (fluence is the number of profiles per unit area).

The particle fluences and doses are additive. The values shown in the figures are for light metals (aluminum, magnesium, etc.) or for organic materials, whichever leads to the higher dose. The effective thickness of material (as used in the figures) is the total area density in grams/cm^2 between the source and the component in question.

Earth and Mars radiation environment is approximately isotropic and the values shown are for a point in space. Those as a function of thickness are for a point within a spherical shell of uniform thickness. The

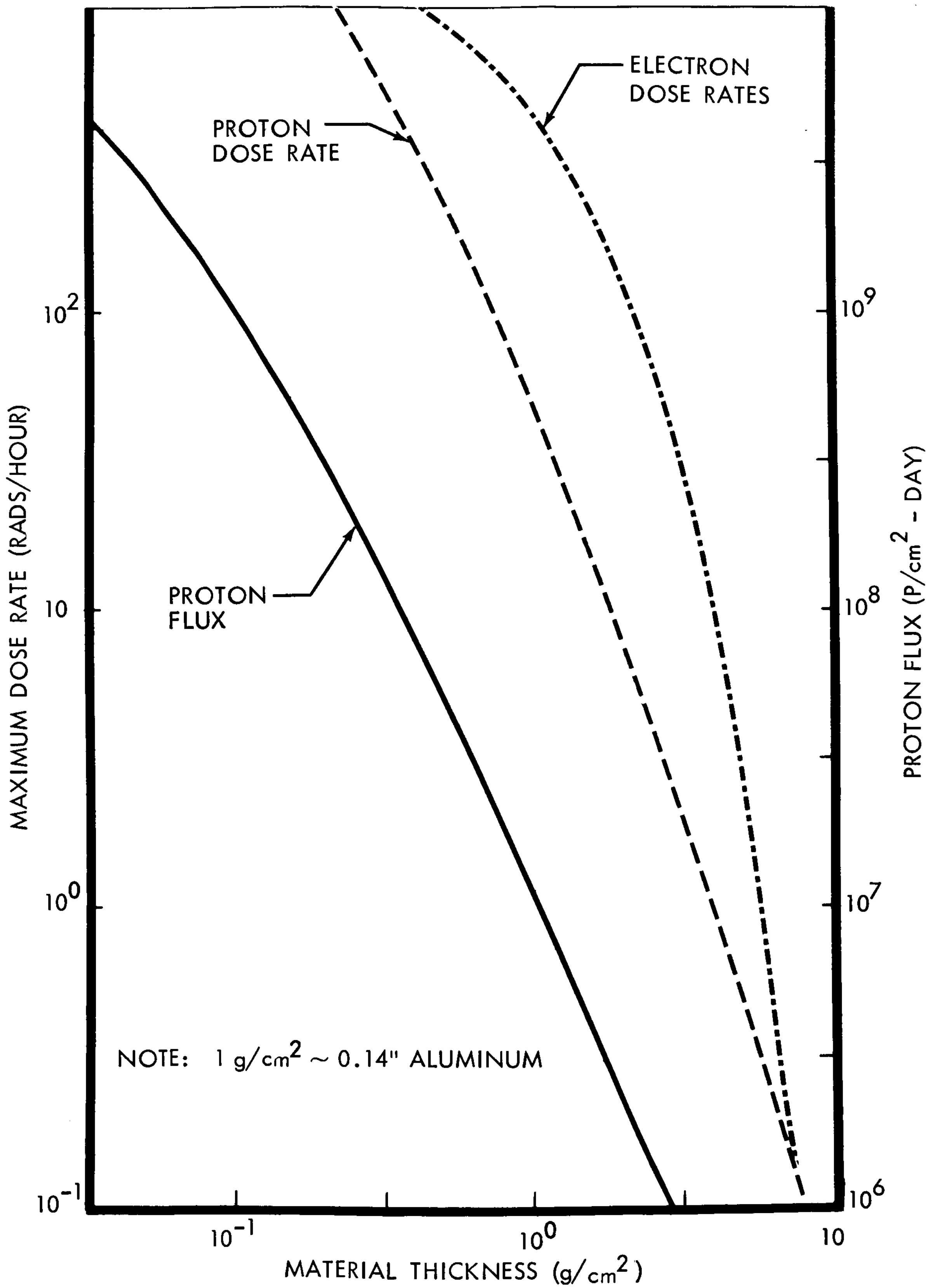
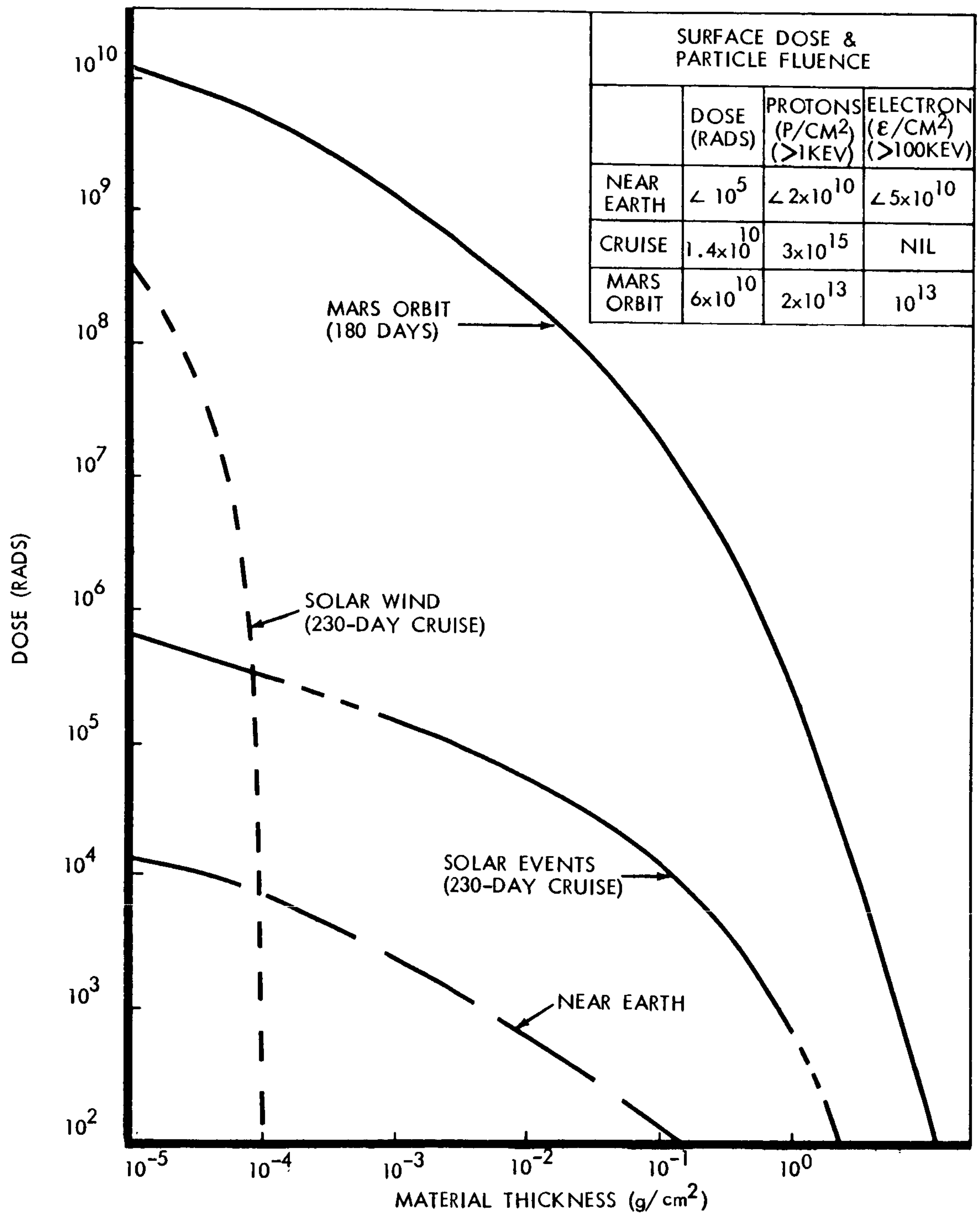


Figure 2-6: Average Proton Flux and Peak Dose Rates Versus Thickness -- Mars Orbit



SURFACE DOSE & PARTICLE FLUENCE			
	DOSE (RADS)	PROTONS (P/CM^2) (>1KEV)	ELECTRON (ε/CM^2) (>100KEV)
NEAR EARTH	$< 10^5$	$< 2 \times 10^{10}$	$< 5 \times 10^{10}$
CRUISE	1.4×10^{10}	3×10^{15}	NIL
MARS ORBIT	6×10^{10}	2×10^{13}	10^{13}

Figure 2-7: Radiation Dose Versus Material Thickness

values in a vehicle must be calculated considering the actual thickness between the component and space. Variations due to thickness should be proportioned according to the solid angle subtended by each sector of a given thickness.

For thicknesses of 1 gram/cm^2 or less, the solar charged particle radiation environment will be considered isotropic. For thicknesses greater than 1 gram/cm^2, the flux will be considered as a collimated beam from the Sun.

The JPL preliminary Voyager specification was used for the solar wind and particle events. However, an exponential rigidity spectrum was fitted to the flux values for determining dose near the surface rather than the power law representation specified by JPL. The specified power law indicates an infinite dose at zero shield thickness. To determine a more realistic depth dose curve, an exponential rigidity spectrum was fitted to the flux values that were specified. The resultant average rigidity of P_o = 87.5 Mv agrees very well with data from Solar Cycle 19 for the year of maximum fluence, 1959.

The dose was evaluated from both the power law and exponential rigidity. The power law representation was used down to 10^{-2} grams/cm^2. Below this, a value was used that was between the two except the surface dose was limited to 10^6 rads as predicted by the exponential rigidity.

2.2.4.7.3 Solar Ultraviolet

Portions of the spacecraft exposed to the Sun will be designed to withstand 4200 equivalent Sun hours (ESH) ultraviolet during cruise and 2400 ESH ultraviolet during Mars orbit. An equivalent Sun hour is the time-integrated intensity (watt-hours/ft^2) of ultraviolet energy corresponding to 1 hour exposure to the Sun at 1 A.U. The solar ultraviolet is defined as that portion of the solar spectrum defined by the Johnson curve with wavelengths of 3800 angstroms of less.

2.2.4.8 The Meteroid Environment

Near Earth--The near-Earth design particle flux is given by:

$$\log N = -17.0 - 1.70 \log M \tag{1}$$

where N = number of particles/(m^2 - sec) of mass M and greater

The density of particles is given by $\rho = 0.4$ gram/cm^3

The velocity of particles is V = 30 km/sec (as given by W.M. Alexander, C.W. McCracken, L. Secretan and O.E. Berg in "Review of Direct Measurements of Interplanetary Dust from Satellites and Probes," a paper presented at the COSPAR meeting, May 1962).

The graph of Equation 1 is shown in Figure 2-8.

Cruise--The near-Earth cruise design particle flux is given by:

$$\log N_E = -13.80 - \log M + 2 \log (0.44/\rho) \tag{2}$$

for $10^{-1.9}$ grams < M

and

$$\log N_E = -14.48 - 1.34 \log M + 2.68 \log (0.44/\rho) \tag{3}$$

for M < $10^{-1.9}$ grams

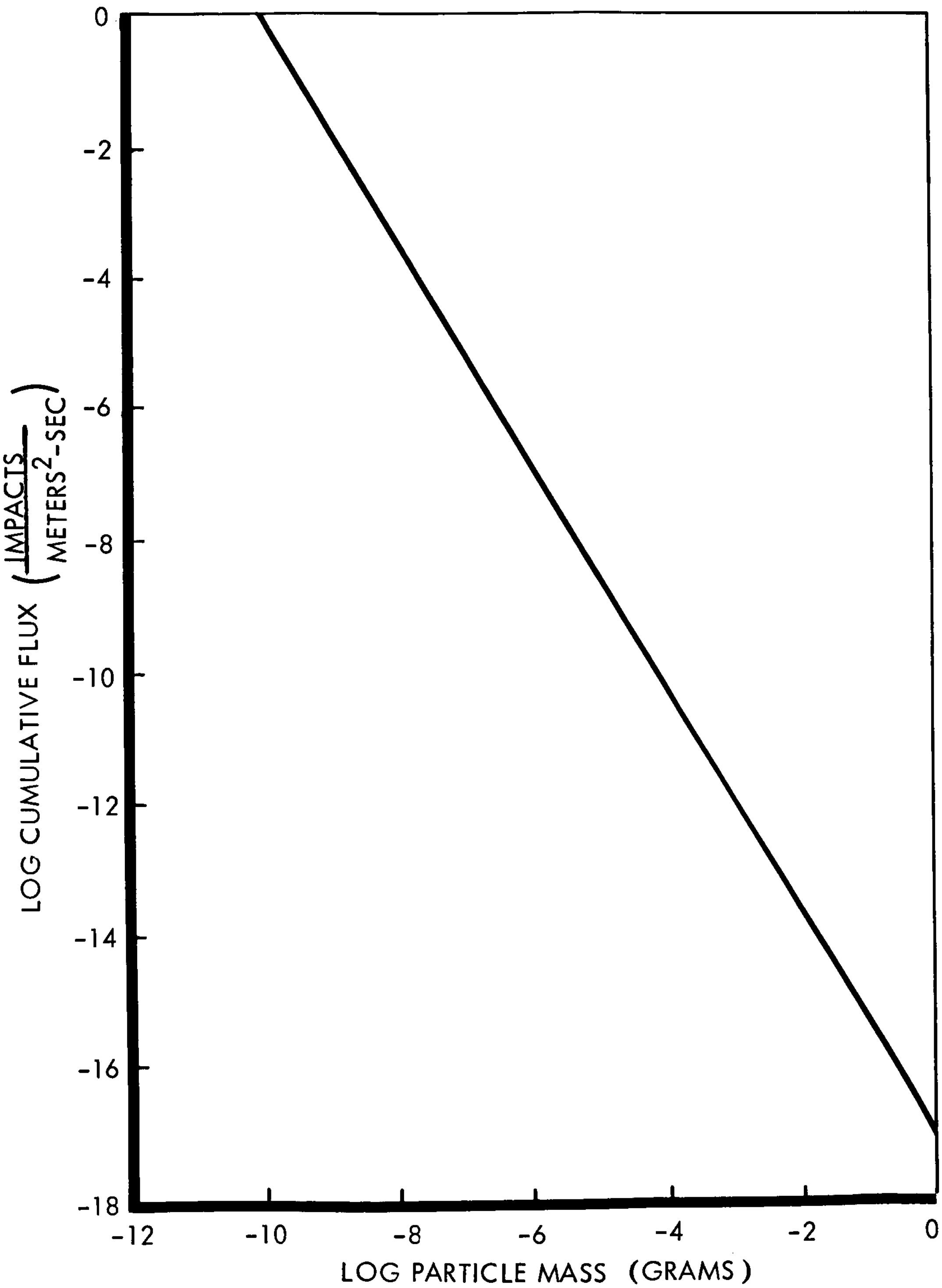


Figure 2-8: Near Earth Meteoroid Environment

where N_E = number of particles/(m^2 - sec) of mass M and greater in the vicinity of the Earth

ρ = 0.4 gram/cm^3

$\bar{V}_{av}$ = 40 km/sec

The near-Mars cruise design particle flux is given by

$$\log N_M = -13.30 - \log M + 2 \log (0.44\rho) \qquad (4)$$

for $M > 10^{-1.9}$ grams

and

$$\log N_M = -13.98 - 1.34 \log M + 2.68 \log (0.44/\rho) \qquad (5)$$

for $M < 10^{-1.9}$ grams

The cruise transit design particle flux is selected as the average of the values obtained from the near-Earth cruise design particle flux (Equations 2 and 3) and the near-Mars cruise design particle flux (Equations 4 and 5). This has the merit of simplicity and is not significantly different from either extreme. The resulting design equations are:

$$\log N_T = -13.45 - \log M + 2 \log (0.44/\rho) \qquad (6)$$

for $M > 10^{-1.9}$ grams

and

$$\log N_T = -14.23 - 1.34 \log M + 2.68 \log (0.44/\rho) \qquad (7)$$

for $M < 10^{-1.9}$ grams

In Equations 2, 3, 4, 5, 6, and 7

N = number of particles/(m^2 - sec) of mass (M) and greater

ρ = 0.4 gram/cm^3

$\bar{V}_{av}$ = 40 km/sec

The subscripts E, T, and M refer to near-Earth cruise, transit cruise and near-Mars cruise, respectively.

The near-Mars cruise design asteroidal flux is given by:

$$\log N_A = -11.7 - \log M \quad (8)$$

where N_A = number of particles/(m^2 - sec) of mass M and greater

ρ_A = 4.37 grams/cm^3

$\bar{V}$ = 24 km/sec

The derivation of Equation 8 is given at the end of this section.

The graphs of Equations 2, 3, 4, 5, 6, 7, and 8 are shown in Figure 2-9.

<u>Near-Mars</u>--The near-Mars design particle flux is given by:

$$\log N = -17.20 - 1.70 \log M \quad (9)$$

for $M < 10^{-5.7}$

and

$$\log N = -13.30 - \log M + 2 \log (0.44/\rho) \quad (10)$$

for $M > 10^{-5.7}$

where N = number of particles/(m^2 - sec) of mass M and greater

ρ = 0.4 gram/cm^3

$\bar{V}_{av}$ = 40 km/sec

The near-Mars design asteroidal flux is given by:

$$\log N_A = -11.7 - \log M \quad (11)$$

(Note: Equations 8 and 11 are identical)

Graphs of Equations 9, 10 and 11 are shown in Figure 2-10.

On the basis of both theoretical considerations (G. S. Hawkins, "Asteroidal Fragments," Astron, J., <u>65</u>, 318, 1960; S. Piotrowski, "The Collisions of Asteroids," Acta Astron, <u>5</u>, 115, 1953) and observational data (C.H. Shuette, "On the Total Mass and Numbers of the Minor Planets," Pop. Astron, <u>58</u>, 438, 1950), the size distribution of asteroidal grains can be

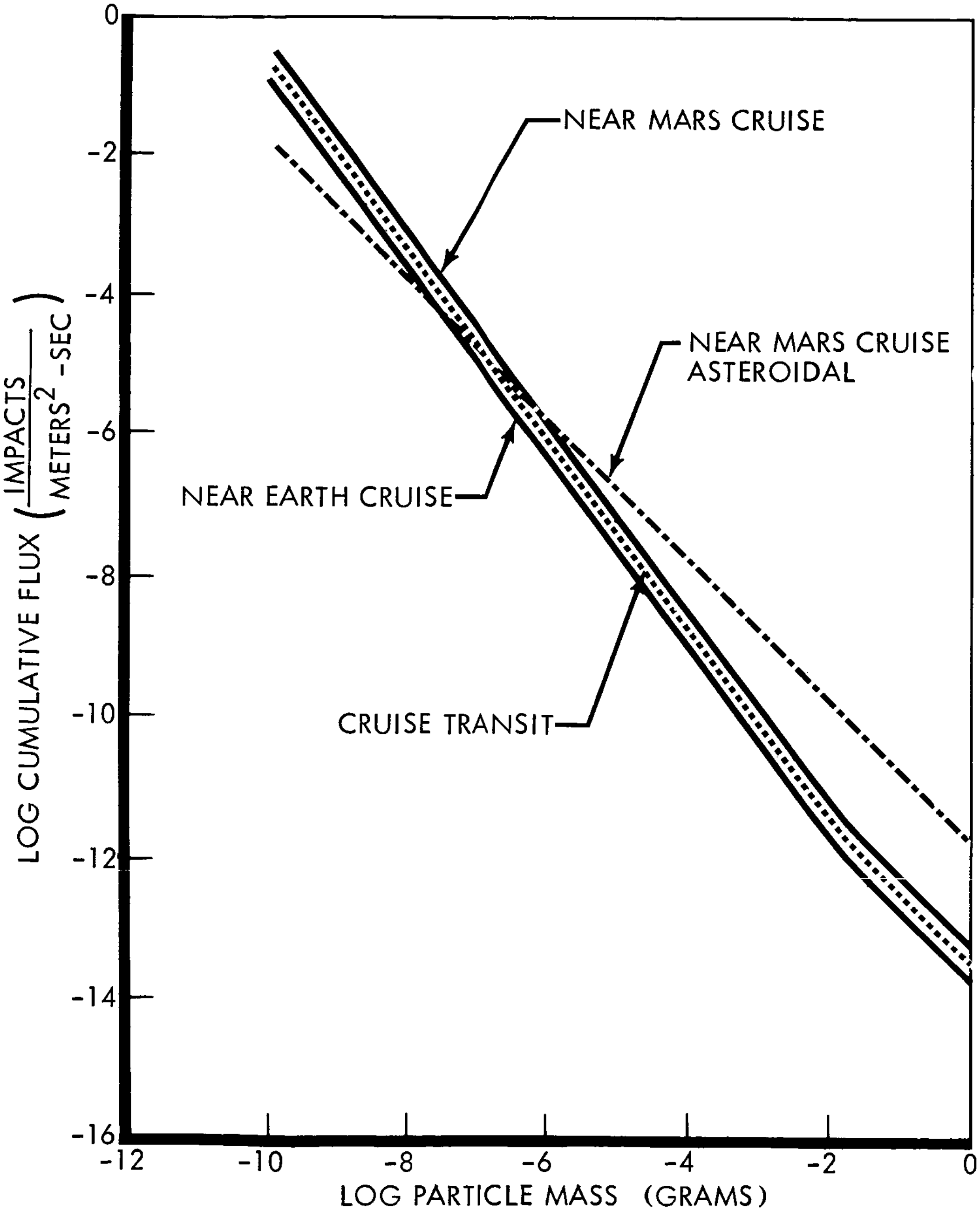


Figure 2-9: Cruise Meteoroid Environment

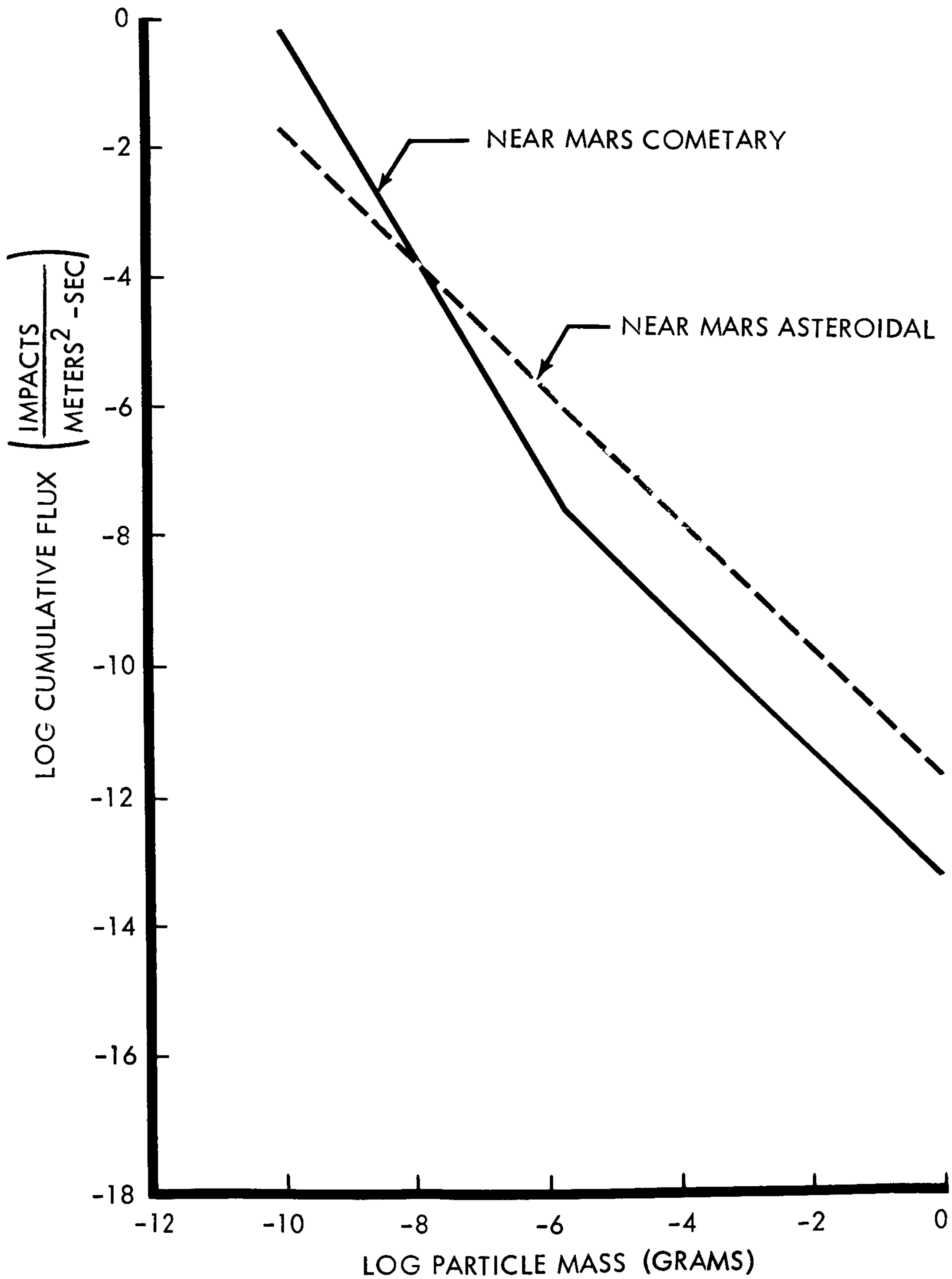


Figure 2-10: Near Mars Meteroid Environment

represented by a log-normal distribution of the form

$$dY = c_o \, a^{-3} \, da, \qquad (12)$$

where dY is the number of particles with radii between a and (a + da). Alternatively, Equation 12 can be expressed as a mass distribution:

$$dY = c_1 \, M^{-5/3} \, dM. \qquad (13)$$

These relations are assumed to be valid over the entire range of asteroidal debris, from the smallest particle retainable under the combined influence of solar gravitation and radiation pressure (the Poynting-Robertson limit).

$$M_o = 4.7 \times 10^{-4} \text{ gm}, \qquad (14)$$

and the largest planetoids (excepting the unique case of Ceres), with masses of the order

$$M_1 = 5.0 \times 10^{22} \text{ gm}. \qquad (15)$$

The foregoing extremes are calculated using an assumed mass density of 4.37 gm/cm^3.

Assuming this variation to be representative of the total population, an upper limit to the density of planetoid grains in the neighborhood of Mars (at 1.52 AU) will be obtained in the debris is assumed to be uniformly distributed throughout the volume V_a of a cylindrical shell, bounded in the radial direction by the orbits of Mars and Jupiter (1.5 and 5.2 AU, respectively) and in the axial direction by planes parallel to the ecliptic, at a distance, h, to either side. An estimate of the distance, h, is given by the product of 2.9 AU (the average distances of asteroids from the Sun) and tan $\bar{i}$, where $\bar{i} \sim 10^o$ is the mean inclination of asteroid orbits, giving h ~ 0.51 AU (C. W. Allen in "Astrophysical Quantities," University of London, Athlone Press, 1955).

The volume, $\bar{V}_a$, turns out be approximately 2.67 x 10^{41} cm^3. With the assumption of a uniform distribution throughout V_a and an average collector speed of $\bar{U}$ ~ 24.14 km/sec, corresponding to a target in orbit about Mars, the differential influx rate dN of asteroidal particles is related to dY of Equation 13 by

$$dN = \frac{\bar{U}}{V_a} \quad dY = \frac{\bar{U}C_1}{V_a} \quad M^{-5/3} \quad dM \tag{16}$$

which can be integrated to give the cumulative influx rate of particles with mass ≥ M grams:

$$\log N = \log\left(3\bar{U}C_1/2\bar{V}_a\right) -0.67 \log M. \tag{17}$$

All that remains is to evaluate the constant C_1 from the total mass M_A of asteroidal material which is given by C. W. Allen as 2 x 10^{24} gm. M_A is represented symbolically by the integral (from M_o to M_1) of MdM so that

$$M_A \sim 3\ c_1\ m_1^{1/3}. \tag{18}$$

giving $c_1 \sim 1.31 \times 10^{16}$. Substituting evaluated quantities into Equation 17 and allowing for gravitational enhancement of the particle density (from D. B. Beard, "Interplanetary Dust Distribution," Astrophysical Journal, Volume 129, 1959), we obtain

$$\log N = -11.7 - 0.67 \log M \tag{19}$$

where N is expressed in particles/(m^2 - sec) and M is in grams. To obtain a limiting case, the coefficient of log M may be increased to -1.0, corresponding to Earth-based observations of the mass distribution of stony meteorites (from G. S. Hawkins). Thus, the upper limit of the cumulative influx rate of asteroidal grains in the near-Mars environment is taken to be

$$\log N = -11.7 - 1.0 \log M. \tag{20}$$

2.2.4.9 Electromagnetic Environment

Near-Earth Magnetic Field (Geomagnetic Field)--The geomagnetic field is evaluated from the magnetic potential

$$V = a\sum_{n=0}^{\infty} \sum_{m=0}^{n} \left(\frac{a}{r}\right)^{n+1} P_n^m (\cos \theta) \quad (g_n^m \cos m\lambda + h_n^m \sin m\lambda)$$

where a is the mean radius of the Earth and r, θ, and λ are spherical coordinates with origin at the Earth's center. The variable, r, is the distance from the Earth's center; θ the geographic colatitude; and λ the geographic longitude. The Gaussian coefficients, g_n^m and h_n^m, are as given by Jensen and Cain ("An Interim Geomagnetic Field" J. Geophys Res 67, 3568, 1962). This representation of the magnetic field represents the geomagnetic field out to 6 to 7 Earth radii. Beyond this point the geomagnetic field becomes distorted by the solar wind. A transition occurs at 11 to 15 Earth radii where the magnetic field becomes erratic and a second transition at approximately 20 Earth radii at the outer boundary of the transition region. Beyond this point the magnetic field is that of interplanetary space.

Cruise Magnetic Field(Interplanetary)--The field strength of the interplanetary magnetic field ranges from 0 to 20 gamma (10^5 gamma = 1 gauss), depending on solar activity in the vicinity of 1.0 AU; it averages about 5 gamma. A maximum upper limit may be as high as 100 gammas

Flyby Magnetic Field (Martian)--A model for the Martian magnetic field based on theoretical calculations would have a large factor of uncertainty. The estimates tend toward upper limits; thus, the martian field strength may be between 10^{-3} and 1 gauss where 10^{-3} is approaching the estimates for the interplanetary magnetic field. Until measured data are available which will allow more definitive estimates of the martian magnetic field, the maximum equatorial field strength in Mars magnetic field is assumed to be half that of the Earth at the same relative altitude.

2.2.5 Planetary Quarantine

The probability of contaminating the planet with any one launch will be no greater than 1 in 10,000. Apportionment of the 1-in-10,000 constraint probability among the contributing events was made on the basis of engineering judgment. Considerations leading to the apportionment are discussed fully in Volume B, Section 3.3. Presented below are the constraint values in relation to the total constraint.

1) Centaur booster impact $P_c = 0.5 \times 10^{-5}$

2) Sterilization canister impact $P_c = 0.5 \times 10^{-5}$

3) Flight capsule contamination $P_c = 3.0 \times 10^{-5}$

4) Flight spacecraft contributions $P_c = 6.0 \times 10^{-5}$

   a) Impact 3×10^{-5}

      (1) At encounter 1×10^{-5}

      (2) From orbit 2×10^{-5}

   b) Contamination by propulsion system products 1×10^{-5}

      (1) Orbit insertion propulsion 0.4×10^{-5}

      (2) Orbit trim propulsion 0.4×10^{-5}

      (3) Attitude control 0.2×10^{-5}

   c) Contamination by meteoroid spalling ejecta 2×10^{-5}

2.2.6 Attitude Control Restraints

The amplitude of spacecraft attitude control limit cycle must be held to ±0.2 degree along each axis to maintain overall pointing accuracy of the high-gain antenna within ±0.6 degree. This restraint may be relaxed during most of the transit portion of the mission to ±0.4 degree attitude control limit cycle.

2.2.7 Vehicle Orientation Restraints

Operation of high-gain antenna (HGA) communication is dependent on vehicle orientation because in some attitudes the vehicle shadows the antenna line of sight to Earth. The HGA hinge axis can point the antenna no closer than 30 degrees to the vehicle negative-Z axis (toward the nose of the shroud). The combination of hinge motion and vehicle roll permits the HGA to cover all of the celestial sphere except the 30-degree cone centered on the negative-Z axis. Within this available field of view, HGA pointing still depends on restricting vehicle roll attitude to certain ranges.

During an early midcourse maneuver, reorientation of the vehicle must be accomplished in a definite sequence to avoid having low-gain antenna radiation pattern nulls point at the Earth. The proposed sequence is: first, rotate the vehicle about its Y axis to align the X axis as closely as possible with the Earth-look line; second, rotate the vehicle about its X axis to align the X-Z plane so that the desired thrust direction lies in the X-Z plane; finally, rotate the vehicle about its Y axis is to align the thrust axis with the desired thrust direction. This procedure allows the Earth-look angle to remain approximately 15 degrees of the maximum of the antenna pattern.

2.2.8 Reliability

The spacecraft design is rigidly disciplined as part of a rigorous reliability program detailed in Section 3.0. The design restraints of that program are briefly summarized.

Redundancy--Within the spacecraft design restraints such as weight, volume, etc., all primary mission functions will be designed with a redundant mode. Insofar as possible, the redundant mode will be commanded, actuated, and performed by equipment not involved in the primary mode of operation, and its elements will have a different failure pattern. Redundant modes providing highly reliable backup will be considered, even at the expense of some reduction in performance. Effective application of redundancy will be assessed using failure mode and effect analysis techniques.

Parts Selection--The parts used in the design will be selected from the Voyager JPL-approved parts list. This list will consist of highly reliable space-proven electronic parts selected from the JPL Sterilizable, Hi-Rel, and Preferred Parts Lists and supplemented with necessary parts of comparable reliability.

Parts Derating--Parts will be applied in the design in accordance with the derating criteria in the Voyager JPL-approved parts list.

<u>Materials Selection</u>--Materials used in the design will be selected from Voyager JPL-approved materials list. Nonmagnetic materials will be used whenever possible.

<u>Processes Selection</u>--Processes specified for the manufacture of the design will be selected from the Voyager JPL-approved processes list.

<u>Design Margin</u>--Adequate margins for tolerance buildup and drift will be provided and assured by appropriate use of worst-case a.c., d.c., or transient analyses.

<u>Thermal Analysis</u>--Adequacy of thermal design of circuits and components will be assured by thermal analyses.

<u>Reliability Assessment</u>--A quantitative numerical assessment will be performed at the component level to determine compliance with the reliability allocation.

<u>Failure Mode, Effect, and Criticality Analyses</u>--During the early design phase, these analyses will be made at the system level and expanded to include the component level.

<u>Design Review</u>--All system, subsystem, and major component designs are critically reviewed at major milestones throughout the design phase.

<u>Subcontracted Design</u>--Subsystem and major component designs will be controlled to the same standards as the contractor designs by rigid

design specifications imposing reliability requirements consistent with the Reliability Program Plan defined in Section 5.0 of this volume. When off-the-shelf designs are used, the design shall include part quality control, proper application of parts, and qualification for the Voyager mission.

Test Requirements--System, subsystem, and component reliability assurance tests shall be specified and performed in accordance with the integrated Test Program designated in Section 7.0 of this volume.

2.2.9 Safety

The spacecraft design is developed in concert with the Safety Program detailed in Section 5.0. The design restraints of that program are briefly summarized.

Electroexplosive Devices (EED)--The electroexplosive design requirements which affect safety are discussed in Section 2.1.3.8 of this volume.

Pyrotechnic Safing--Switches will be incorporated to maintain pyrotechnic equipement in a safe condition while activation would be hazardous to personnel or equipment. Unlatching devices will be protected against spurious signal unlatch.

Pressure Vessels--Vessels hazardous to personnel shall be designed with adequate safety margins. Each design will be analyzed, potential hazards identified, and specific safety requirements set as discussed in the Safety Plan in Section 5.0. Safety Margins of 2.2 for hazardous pressure vessels and 1.15 for rocket motor cases shall be considered a normal minimum. Vessels with wall thickness-to-diameter ratios smaller than 1/1000 shall be avoided.

Fault Analysis--The safety of the design will be evaluated and potential critical fault paths identified using the "Fault Tree Analysis" technique discussed in the Safety Plan in Section 5.0

Design Review--Design reviews conducted throughout the design phase as part of the Reliability Program will be utilized to assure the integration of the safety requirements into the design.

2.3 GUIDANCE AND NAVIGATION--MANEUVER ACCURACY AND PROPULSION REQUIREMENTS

The maneuver accuracy requirements for all thrusting periods after trans-Mars injection and the Δv requirements for midcourse maneuvers and orbit trim are presented in this section. The analysis leading to these requirements is presented in Volume B, Subsection 3.1, and the detailed allocation of the accuracy requirements to component error sources (gyro drift, accelerometer bias, thrust vector control, etc.) is presented in Volume A, Subsection 3.8.

2.3.1 Maneuver Accuracy Requirements

For all maneuvers, it is desired to provide total control accuracies as follows:

1σ pointing error $\leq$	0.010 radian
1σ ΔV proportional error $\leq$	1.0 percent of Δv
1σ ΔV resolution error $\leq$	0.01 meters per second for midcourse corrections and orbit trim
	$\leq$ 4.5 meters per second for orbit insertion

These accuracies allow control of the Mars encounter to 500 kilometers, and control of final orbit periapsis to $3\sigma P \leq 5$ kilometer, and semimajor axis to $3\sigma a \leq 30$ kilometers.

2.3.2 Midcourse Maneuver Δv Requirements

The Δv required for correction of trajectory errors due to random injection and maneuver execution errors is 48 meters per second. This requirement is based on a nominal midcourse correction sequence which includes three corrections--one at 5 days, one at 25 days, and one at 175 days after launch. It is deemed desirable to add Δv to this requirement to allow correction of nonabortive failures during trans-Mars injection and midcourse maneuvers. Although the latter is not an easily defined quantity, because no Centaur data are available from which to make estimates, an increase of 50 percent in total Δv requirements is a logical estimate. Also, a small additional Δv is required for aim-point biasing. The total midcourse Δv requirements is then about 75 meters per second.

2.3.3. Orbit Trim Δv Requirements

Orbit trim is assumed to take place in two steps. The first will be a trim maneuver at orbit apoapsis and will take place after the first few orbits which will be allotted to orbit determination. This maneuver will require about 15 meters per second for adjustment of periapsis altitude. After the first maneuver, the uncontrolled orbit parameters will be slightly perturbed, therefore, a few orbits must be allowed for orbit redetermination. The second trim maneuver will then be executed at periapsis to control the semimajor axis. This maneuver will require another 75 meters per second. A conservative estimate, therefore, of the orbit trim Δv requirement including a small allotment for changing the nominal orbit will be 100 meters per second.

2.4 AIMING POINT SELECTION

2.4.1 Scope

This section describes relevant capabilities and constraints for systems, spacecraft subsystems, and scientific experiments that influence the aiming point selection for the 1971 Voyager mission. A procedure is illustrated by which these constraints are considered in the selection of an aiming point.

2.4.2 Aiming Point Description

The aiming point is specified by two components of the impact parameter, $\bar{B}$. The impact parameter is a vector perpendicular to $\bar{S}$, which is in the direction of the approach asymptote to the planet. $\bar{B}$ has a magnitude equal to the distance from the planet center to the asymptote of the approach hyperbola. The two components of $\bar{B}$ used to specify the aiming point are the projections along the $\bar{T}$ and $\bar{R}$ unit vectors. $\bar{T}$ is a unit vector perpendicular to $\bar{S}$ and parallel to the ecliptic, while $\bar{R} = \bar{S} \times \bar{T}$. The aiming point is then specified by the components $B \cdot T$ and $B \cdot R$, or by the magnitude $b = |\bar{B}|$ and the direction θ, where θ is the angle between $\bar{T}$ and $\bar{B}$ measured positive in the clockwise sense in the $\bar{R}$ - $\bar{T}$ plane ($\bar{S}$ into the $\bar{R}$ - $\bar{T}$ plane). See Figure 2.4-1.

It should be noted that the choice of an aiming point does not significantly affect the heliocentric trajectory. The direction of $\bar{S}$ is a function of launch date and arrival date, and is essentially the same for all aiming points in the vicinity of Mars. The asymptotic velocity V_{∞}, is similarly unaffected by the choice of aiming point. On the other hand,

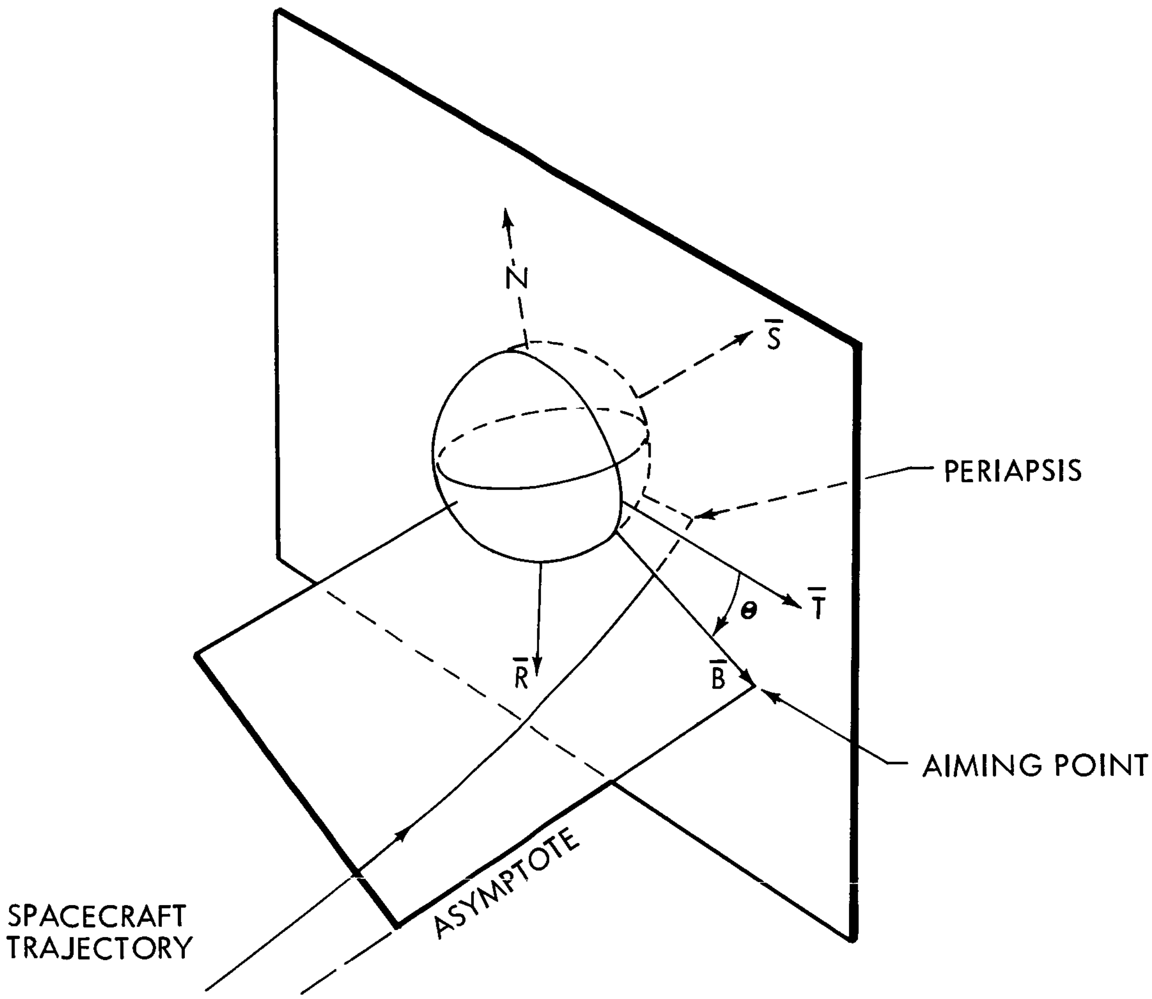


Figure 2.4.-1: Aiming Point Geometry

for a fixed $\bar{S}$ and V_∞ variations in the Mars-centered approach hyperbola are directly related to changes in $\bar{B}$.

2.4.3 Mission Objectives

The mission objectives affect the aiming point selection by specifying the relative weightings given to the various scientific experiments. The most significant influence of these weightings is in the selection of the inclination of the orbit about Mars, which in turn specifies the aiming angle, Θ.

2.4.4 Design Characteristics and Constraints

2.4.4.1 Systems Considerations

Planetary Quarantine--The apportioned probability that the Flight Capsule will contaminate Mars is 3×10^{-5}. Contamination probabilities of 0.5×10^{-5} have been assigned to the Centaur and the sterilization canister, leaving a probability of 6×10^{-5} that the Flight Spacecraft will contaminate the planet, (See Section 3.3 Volume B). Revision of the allocations would affect the aiming point, because a different biasing distance would be required to achieve a satisfactorily low probability of impact of the Flight Spacecraft.

Flight Spacecraft--The above-mentioned planetary quarantine requirement places severe restrictions on the aiming point of an unsterilized spacecraft. The requirement that the orbit about Mars have at least a 50-year lifetime defines the minimum periapsis altitude (and minimum $|\bar{B}|$) for an orbit with a given period. The aiming point must then be selected so that the probability of arriving at a lower periapsis is less than 3 x 10^{-5}. The remaining 3 x 10^{-5} probability of contamination due to the Flight Spacecraft is assigned to contamination while in orbit. Such contamination would be due to propulsion system emissions, particles knocked off by micrometeoroid impact, etc.

Launch Vehicle--The Launch Vehicle does not substantially affect the choice of the final aiming point. Dispersions remaining after correction of injection errors, and biasing of the initial aiming point to prevent Centaur impact, are covered in the discussion of guidance in Section 3.1.4.2. of Volume B.

Deep Space Instrumentation Facility--Insertion into the orbit about Mars, and capsule entry, descent, and landing, will occur in view of the Goldstone DSS. Alternately, time phasing of periapsis (and landing) with desired surface features is available by aim-point selection at insertion to the cruise phase. Approximately one-third of the possible choices can be viewed from Goldstone.

Space Flight Operations--To permit a maximum time for transmission of commands from the Goldstone DSS just prior to orbit insertion, the insertion maneuver is planned to occur during the latter half of the Goldstone viewing period.

2.4.4.2 Approach Trajectories

The choice of aiming point is influenced by both the characteristics of the orbit about Mars and the type of approach trajectory used. In one type, the nominal aimed periapsis altitude is selected so that there is less than a 3×10^{-5} probability that the actual periapsis will be below some minimum altitude. This minimum altitude is the one that just gives a 50-year lifetime for the desired orbit period. This lifetime is computed in a conservative manner, so that there is essentially no probability that the orbit will have less than a 50-year lifetime if its periapsis is above the calculated altitude. With this aiming philosophy, orbit insertion is aborted only if the angular orientation of the spacecraft is unsatisfactory.

Another approach mode uses postmidcourse tracking to trade assurance of orbit insertion for a lower periapsis, which is advantageous to the scientific mission. In this case the aim point is selected such that there is some probability greater than 3×10^{-5} that the guidance dispersions place the periapsis below that of an orbit for compliance with the planetary quarantine constraints. In this mode, tracking data would indicate whether the periapsis altitude is going to be high enough to provide the required 50-year orbit life. In those cases where periapsis altitude is not high enough, the orbital insertion manueuver would be aborted. For the present Voyager analysis, however, estimates of tracking dispersions and guidance dispersions are of such a magnitude that this method has little advantage. For this reason the insertion maneuver and aiming point are planned so that the decision to insert is not dependent upon tracking data.

The use of a solid propellant for the orbit-insertion propulsion system creates a strong desire for a constant ΔV insertion throughout the launch period. This can be provided by using a periapsis insertion mode for trajectories giving a constant V_{∞} as discussed in Section 3.1.2.3. In this mode the orbit periapsis nearly equals the hyperbolic approach periapsis. (The word "nearly" is used to account for small differences due to finite thrust, insertion guidance, etc.)

Constant ΔV insertion can also be obtained by designing the approach trajectory and insertion mode to accommodate varying V_{∞}. This requires an inefficient insertion mode at the lower V_{∞}. There are two ways to accomplish this. One is the "intermediate-orbit" mode, in which the spacecraft is initially placed into an orbit with apoapsis nearly equal to that of the desired final orbit, but with periapsis a few hundred kilometers higher than the desired final value. A subsequent orbit trim maneuver near apoapsis is used to adjust the periapsis altitude. The principal drawback to this mode is the reliance on the orbit trim maneuver to achieve an orbit which is favorable for the scientific experiments.

Another way of obtaining constant ΔV for varying V_{∞} is the "nonperiapsis" insertion" mode, in which the excess ΔV capability is expended in the inefficient insertion. This causes the position of periapsis to be moved from the location it would have had for periapsis insertion. This affects the illumination at periapsis and could be favorable or unfavorable, depending upon the geometry for the particular trajectory. A disadvantage of this mode is that considerable tracking is required for computation of the time of insertion, and there may not be a sufficient time interval after the last midcourse correction to accomplish this. Both of these alternate approach and insertion modes require a higher periapsis altitude for the approach hyperbola than is desired for the orbit about Mars. Therefore, the magnitude of $\overline{B}$ would be greater than that for the periapsis insertion mode.

Using an impact trajectory for the Planetary Vehicle, and deflecting the Flight Spacecraft after separation of the Flight Capsule, has an excessive probability of contaminating Mars unless the spacecraft is sterilized. Even if the spacecraft is sterile, this approach mode offers little increase in landing-site accuracy, and the extra maneuver degrades the probability of achieving a successful orbit.

2.4.4.3 Flight Spacecraft Science Subsystem Consideration

The spacecraft scientific instruments indirectly influence the selection of the aiming point through their influence on the orbit about Mars. Most of the experiments benefit from a comparatively low altitude, so the aiming point is chosen as low as the planetary quarantine requirement allows. The aiming angle, Θ, (See Figure 2.4-1) is chosen to provide the desired orbit inclination. The choice of inclination is covered in Section 3.1.4.

2.4.4.4 Telecommunications

The spacecraft is continuously in view of Earth during the approach to Mars and through orbit insertion. During capsule entry, descent, and landing, the spacecraft will be able to communicate with the capsule and with Goldstone. The aiming angles (θ) that satisfy this constraint are indicated in Section 2.4.5.

2.4.5 Selected Aiming Points

The magnitude of $|\bar{B}|$ is chosen so that the probability of having a periapsis altitude too low for a 50-year orbit lifetime is less than 3 x 10^{-5}. For example, the minimum periapsis altitude is 2000 kilometers for an 18-hour orbit. For this periapsis altitude and a V_∞ of 3.5 kilometers per second, the $\bar{B}$ is 8145 kilometers. However, meeting the planetary quarantine constraint (3 x 10^{-5} probability of contamination) and allowing for guidance dispersions requires aiming for a larger $|\bar{B}|$. With an assumed 1σ dispersion of $\pm$ 167 kilometers, a $|\bar{B}|$ of at least 8814 kilometers must be used for the aiming point.

Because most of the scientific experiments benefit from low periapsis altitudes, the minimum of 8814 kilometers is chosen to be the aimed magnitude of $\bar{B}$ for this example. Considering another example, a 9-hour orbit similarly requires a minimum periapsis altitude of 3100 kilometers, and has a desired $\bar{B}$ of 10,170 kilometers.

The aiming angle θ, which describes the direction of $\bar{B}$, is a prime consideration in the choice of orbit inclination. Earth, Canopus and Sun occultation periods place constraints on Θ and outline a region of acceptable aiming points. This limits the choices of inclination. Figure 2.4-2 shows how these constraints are applied for a representative example. The particular trajectory used as an example is one in which the Planetary Vehicle is launched on April 30, 1971, and arrives on October 31, 1971, with a V_∞ of 3.527 kilometers per second. Various regions are outlined to illustrate the effects of the constraints. Region A is unacceptable because $|\bar{B}|$ is less than the 8814 kilometers minimum described in the previous paragraph. Region B is unacceptable because Earth occultation occurs before encounter, i.e., continuous communication cannot be maintained during the approach to Mars. Region C is unacceptable because at some point on the approach hyperbola some part of Mars comes within 10 degrees of the spacecraft-Sun line. This is close enough to cause loss of attitude reference during a critical phase of the mission. The two regions, D and E, are shown to account for the possibility of Mars entering the rectangular field of view of the Canopus sensor. Region D corresponds to a 35-degree conical field of view and Region E corresponds to a 60-degree conical field of view.

An aiming point within Region D is unacceptable because Mars certainly enters the field of view. With an aiming point within Region E, the orientation of the Canopus Sensor determines whether or not Mars enters the sensor rectangular field of view. This artifice is used because Boeing's quick-look computer program has not yet been updated to account for rectangular fields of view. This is of small consequence for the

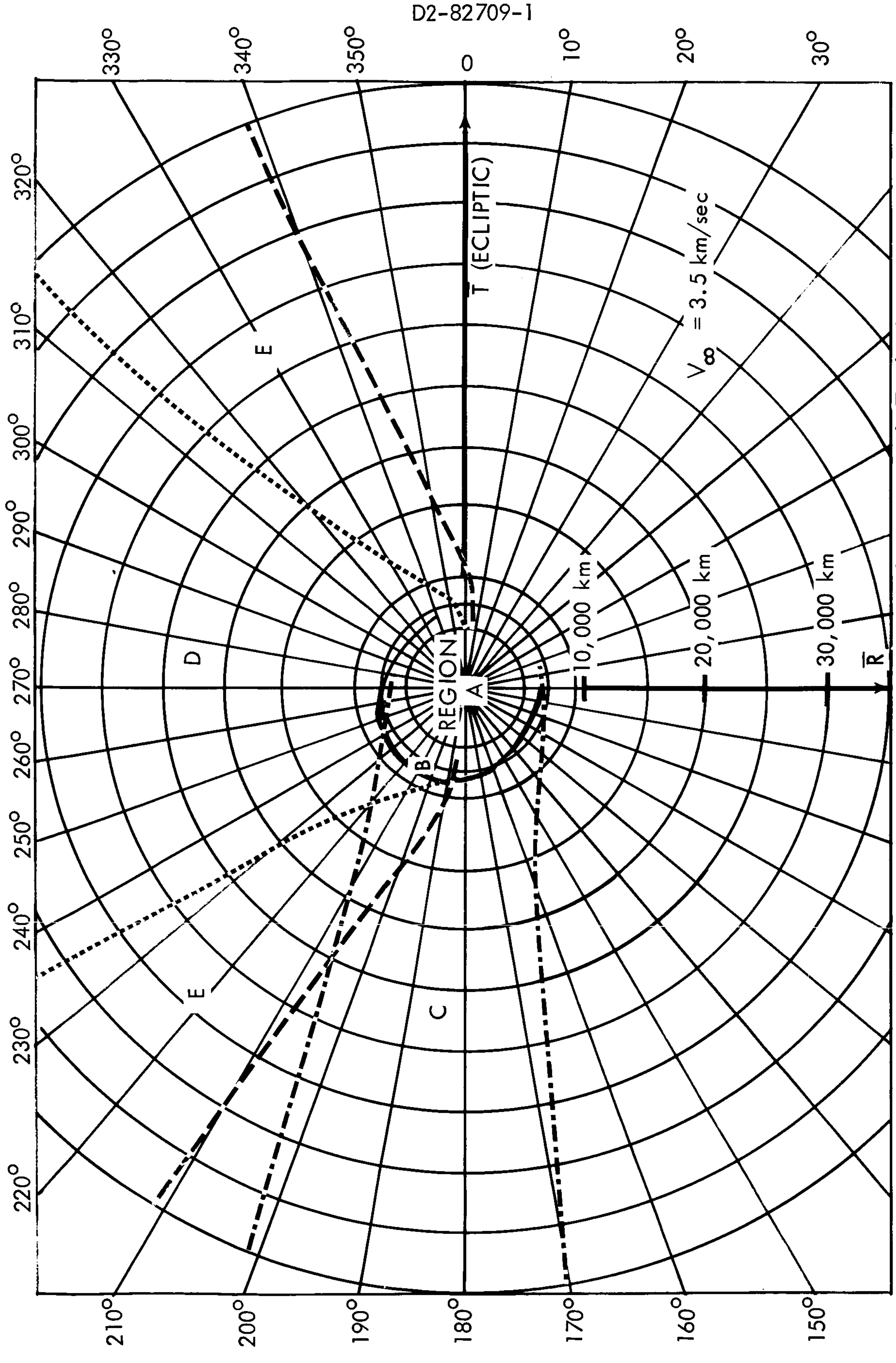


Figure 2.4-2: Voyager Mars 1971 Nominal Transit Trajectory Occultation Before Periapsis — Orbiter

orbiter mission study, however, since this computer program is one of the preliminary steps in the orbital selection process. The critical decisions are made after plotting the cone and clock angle of the center and the limbs of Mars on a point by point basis throughout each candidate orbit. On such a plot, the rectangular Canopus sensor field of view is represented so the times and duration of occultation, if any, are considered. This chart changes only slightly with other transit trajectories of interest. Figure 2.4-3 shows similar data for trajectory launched nine days later and arriving 45 days later, with a V_∞ of approximately 2.85 kilometers per second.

After the constraints on $\overline{B}$ and Θ are identified (as shown in the examples) the final selection of Θ depends upon the orbit inclination desired from scientific data consideration. Figure 2.4-4 shows the relationship between Θ and orbit inclination for the example trajectory. As an example, when the desired orbit inclination, chosen from considerations of occultation and of science, is 40 degrees, the necessary Θ is 15.5 degrees. Other aiming points within the acceptable region could be chosen, however, and inclinations would range from 32 to 121 degrees.

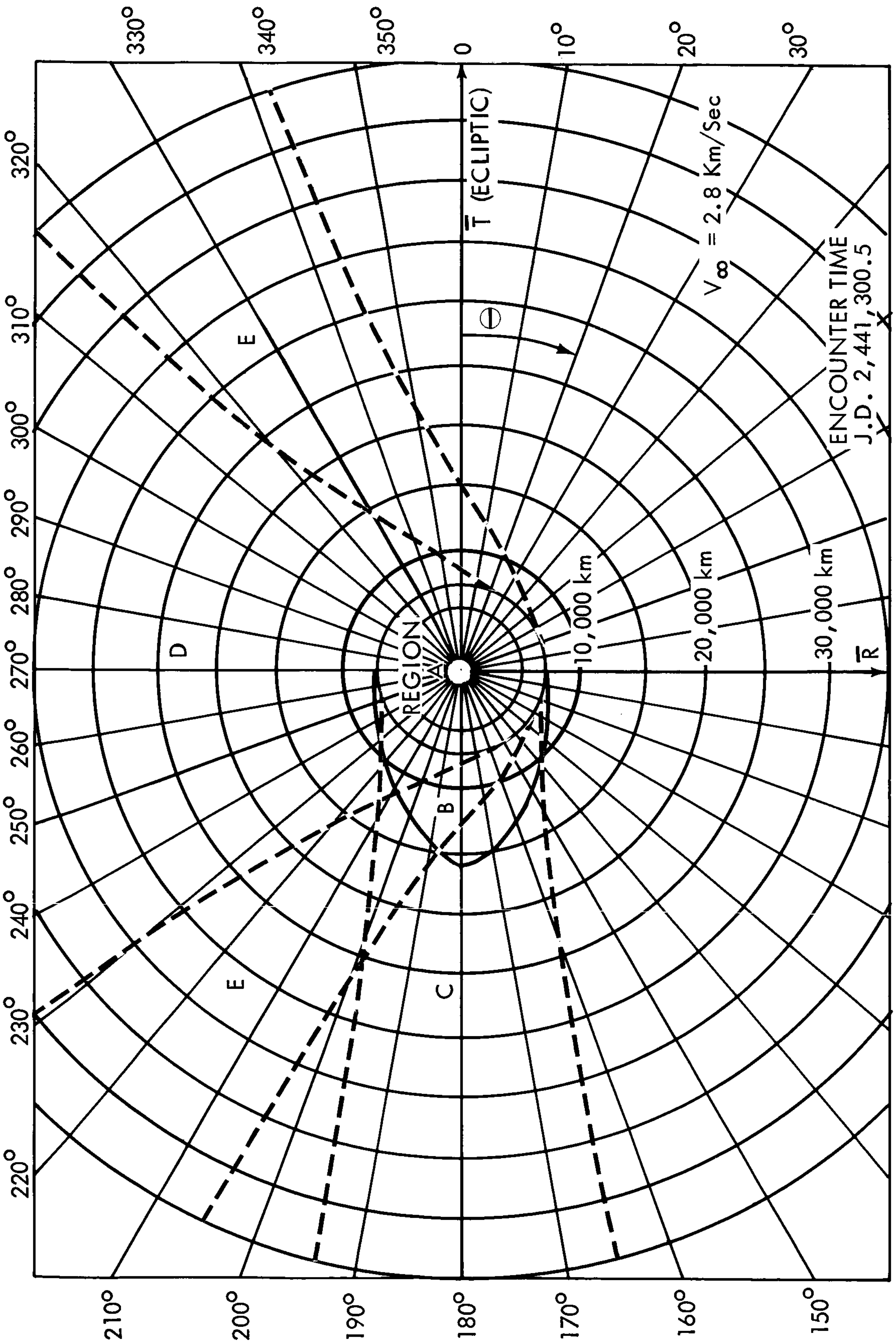


Figure 2.4-3: Voyager Mars 1971 Nominal Transit Trajectory 2 — Occultation Before Periapsis (Orbiter)

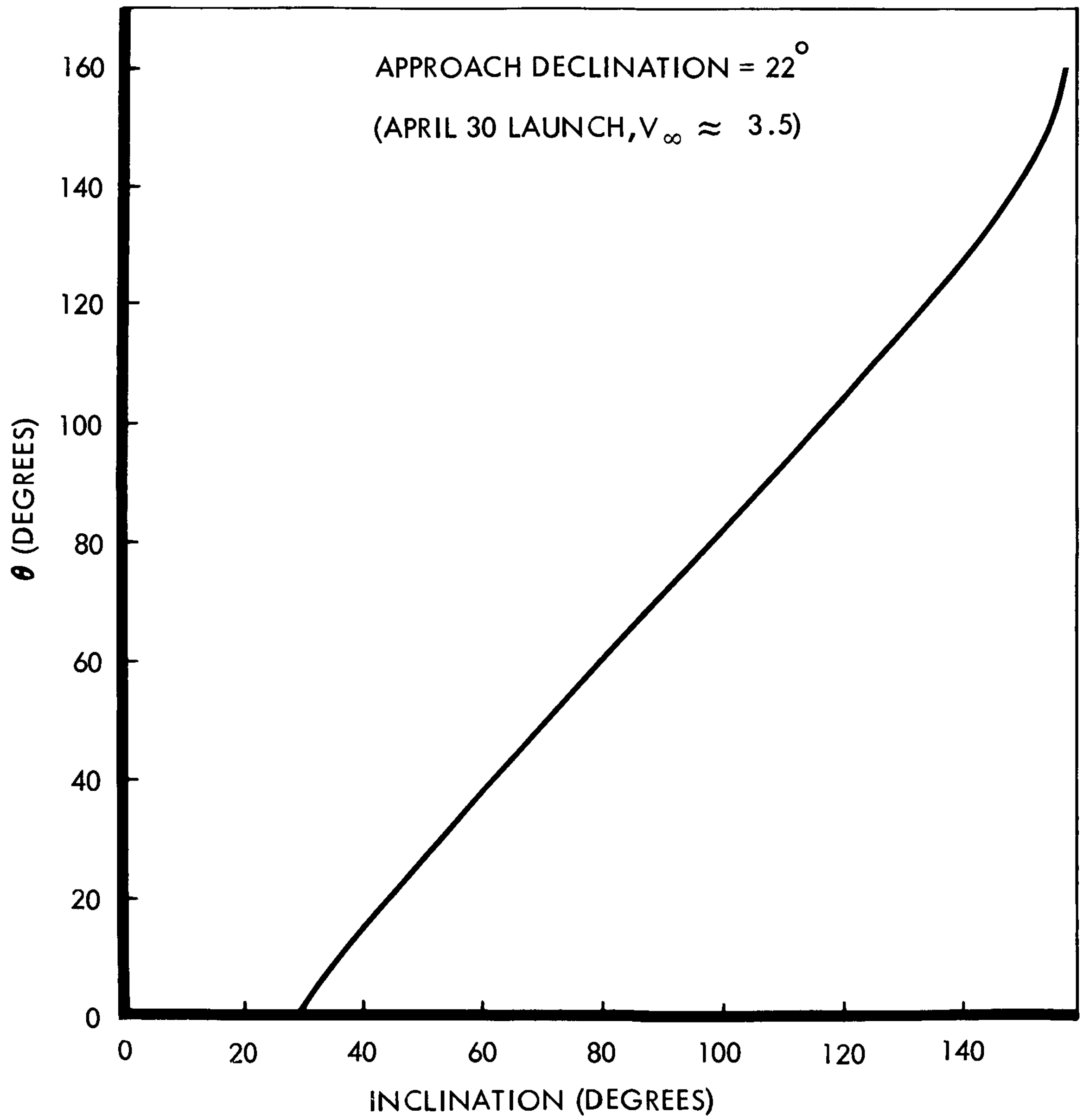


Figure 2.4-4: Available Orbit Inclinations

CONTENTS

3.0 SYSTEM LEVEL FUNCTIONAL DESCRIPTION OF FLIGHT SPACECRAFT

System aspects of the Voyager Spacecraft development are covered in this section. The trajectory choices are discussed in terms of mission versatility subject to constraints on planetary contamination and on desired photographic data; treatment of orbit determination errors indicates the types of errors to be expected and their variances. Design parameters for components of the Flight Spacecraft are discussed, following which the nomenclature for identification of such components (and related documentation) is explained. Requirements on the interfaces between various items of flight equipment and the launch vehicle are described; including consideration of spacecraft separation, dynamic interactions during flight, and launch vehicle performance. Telemetry criteria, including considerations of data formatting, are discussed and a list of flight equipment telemetry channels required for the Voyager system is provided. Guidance and navigation maneuver errors are discussed as to sources of error and variances. The Voyager Flight Sequence is delineated not only in general terms but in terms of details of operation of various items of flight equipment. The various features of the spacecraft layout, configuration, and flight equipment are discussed; the discussion includes spacecraft reference axes, mechanical alignment provisions, general arrangement of the exterior, and provisions for equipment mounting. Considerations of planetary quarantine and their potential effects on vehicle design are discussed; requirements for flight equipment cleanliness are treated; and the requirement for magnetic cleanliness is discussed.

3.1 VOYAGER STANDARD TRAJECTORIES

3.1.1 Scope and Summary

This section describes a variety of trajectories that are applicable to the 1971 mission. The capability of the preferred spacecraft design is such that a number of trajectories and orbits for the 1971 mission can be performed. In addition, the spacecraft capability affords considerable versatility in performing missions in the 1973 through 1977 opportunities as well as for the 1969 opportunity for the test flight.

The spacecraft can enter biologically safe orbits with periods as low as 18 hours from approach velocities (V_∞ at Mars, as high as 3.5 kilometers per second, or with periods less than 9 hours from approach velocities as high as 3.0 kilometers per second. The 18 hour example provides coverage of four different swaths of Mars surface in the first 3 days after encounter. For the 3.5 kilometer per second approach velocity, encounter can occur when the annual Mars wave of darkening has its maximum contrast. At these early arrival dates, orbital periods greater than 18 hours can also be selected. Alternatively, in the interest of obtaining more photographic data (at slightly lower quality), lower orbit periods can be obtained for later arrival dates. For example, the orbits at periods less than 9 hours can be established at arrival dates in the medium contrast time of the wave of darkening where V_∞ = 3.0 kilometers per second. Such lower orbits must have slightly higher periapsis altitudes, but they repeat their passage more often, taking and transmitting more photographic data during the orbiting phase of the mission.

In 1973 missions, the Type I transit trajectories typically have a short launch opportunity. The designed ability to accommodate Mars approach velocities as high as 3.5 kilometers per second allows a 37-day launch opportunity as compared to the 26-day launch opportunity of nearly mass optimized trajectory sets.

Although the present mission plans do not include it, the option exists of performing a similar orbital mission in 1975 over a relatively wide range of arrival dates (on the order of 100 days), or in 1977, if Type II transfers to Mars are used these years.

In 1971, orbits are available that have no occultation of Canopus or the Sun for the first 60 days in orbit. The periapsis positions are at southern latitudes and at illumination angles that favor the black-and-white TV experiment. Some adjustment of periapsis position is available at insertion by off-periapsis orbit insertion. Additional impulse reserve for such an adjustment is obtained by choosing slightly later arrival dates with the present design.

In this section, details of the missions are described through selected examples from the available sets of trajectories.

3.1.2 Transit Trajectories

3.1.2.1 Transit Trajectory Design Criteria

Only Type I trajectories are considered for the 1971 transit trajectory. They are characterized by shorter transit times (compared to the Type II trajectories) and lower communication distances. The transit trajectories are selected to satisfy the operational constraints imposed by booster payload limitations, DSN tracking capability, range safety, and spacecraft propulsion capability. Figure 3.1-1 shows a basic trajectory design chart of geocentric launch energy parameter (C_3) as a function of launch and arrival dates. The applicable constraints are indicated.

The transit trajectories, which satisfy the operational constraints, are evaluated in terms of their effect on communication distance, ovservation of the Mars wave of darkening phenomenon (a progressive albedo decline that proceeds from the polar ice cap toward the equator), target illumination, transit time and compatibility with future missions. Figure 3.1-2 shows some of these effects. The blue curves show the illumination angle at periapsis, measured from the terminator, for Mars orbit inclinations of 40 and 60 degrees. The areas bounded by the red curves show three "seasons" of progressive contrast for the wave of darkening for latitudes from -60 to +20 degrees. The absissa for these curves is latitude rather than launch date. Two arrival philosophies are considered for the transit trajectories: (1) constant arrival date and (2) constant approach speed at Mars. Each concept is discussed for its relative advantages and disadvantages.

3.1.2.2 Constant Arrival Date

The selection of a constant arrival date fixes the communication distance at encounter. It also simplifies the scheduling of the DSN. It has the disadvantage of a variable approach speed at Mars (V_∞) and a shortening of the launch period for a given allowable geocentric energy.

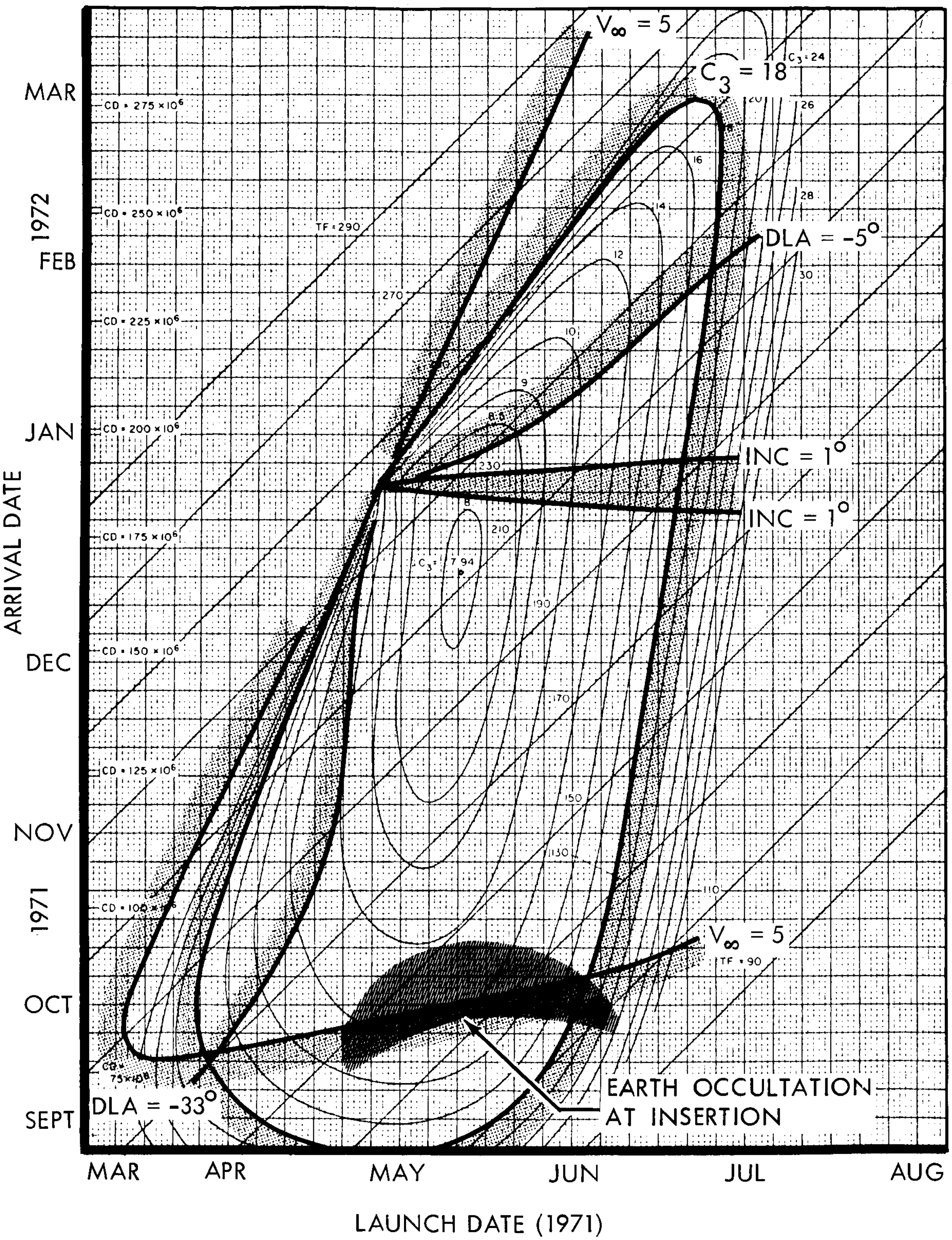


Figure 3.1-1: Trajectory Design Chart

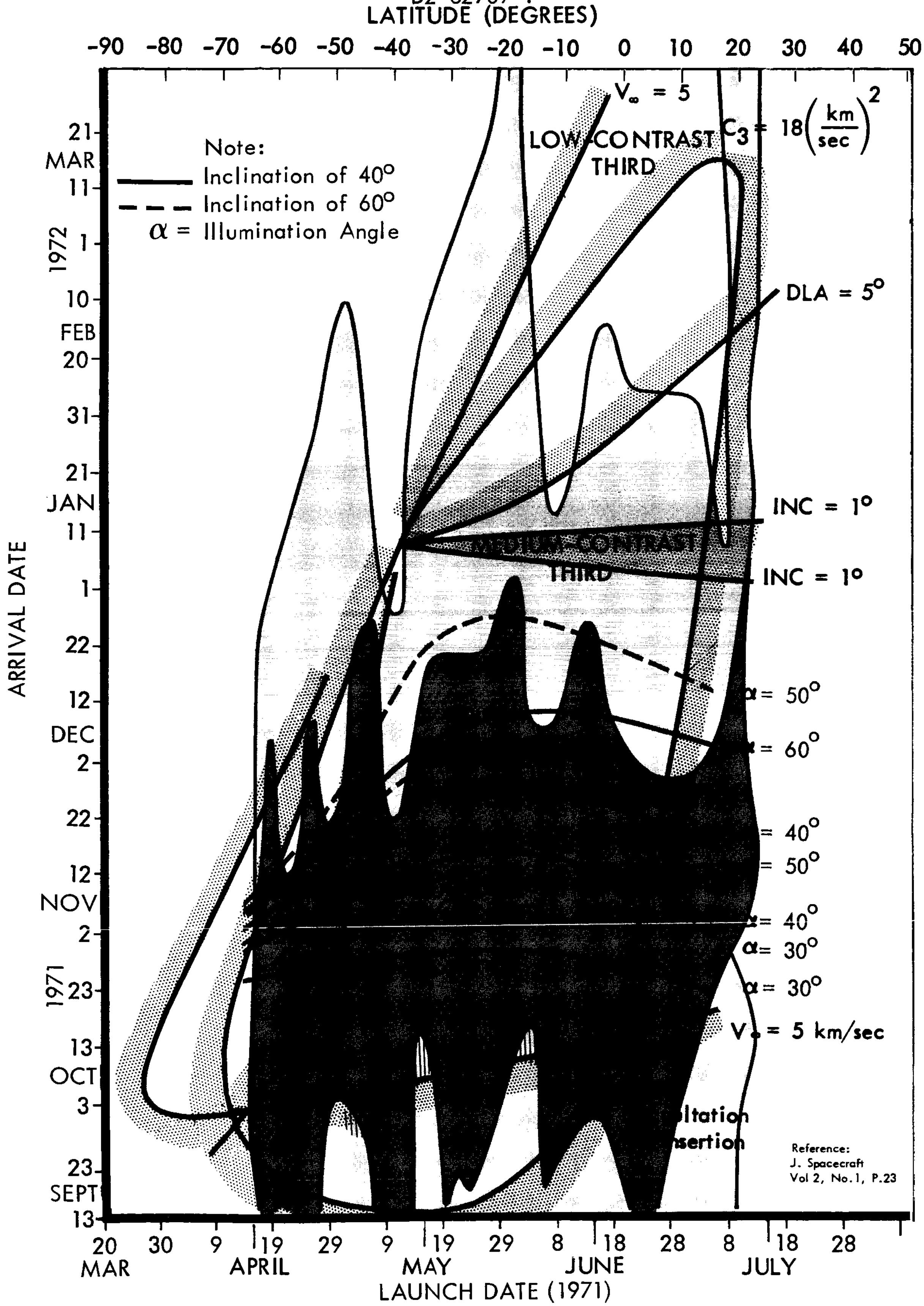


Figure 3.1-2: Transfer Trajectory Constraints

Three representative arrival dates have been selected in the region of interest: October 18, November 25 and December 25, 1971. Each arrival date has been evaluated to determine its advantages and disadvantages. Table 3.1-1 shows a comparison between the trajectories for the above fixed arrival dates. The comparison is made on the basis of maximum launch period, communication distance, observation of the wave of darkening, illumination angle, transit time, and compatibility with future missions. The maximum launch period is the time between the limits of launch azimuth and C_3. All of the selected trajectories have launch periods within the specified 45- to 60-day range. Communication distance at arrival favors the earlier arrival dates; 46 percent less for October 18 than for December 25. Observation of the wave of darkening also favors the earlier arrival dates. Arrival on October 18 would allow viewing in the season of maximum contrast during the first orbits and viewing at a time of maximum rate of change 1-1/2 months later. The illumination angles favor the October 18 or November 25 arrival dates for black and white television. Transit times for the early arrival dates will be shorter implying somewhat greater reliability. Compatibility with the 1973 mission can be inferred from the trajectory design chart, Figure 3.1-3. Arrival date selection for the 1973 mission is limited by DLA and C_3 constraints. A V_∞ of 3.45 is obtained during the maximum launch period with a constant arrival date philosophy. The difference in V_∞ for maximum launch period has been adopted as the criterion for comparison of the 1973 and 1971 orbiter missions. Propulsion compatibility with the 1973 mission favors intermediate arrival dates.

Table 3.1-1: CONSTANT ARRIVAL DATE TRAJECTORY SETS

Arrival Date	Max. Launch Period, Days	Communication Distance at Arrival (km)	Obs. of Wave of Darkening at Arrival	Illumination Angle (degrees)	Transit Time (Days)	*Difference Between 1971 and 1973 V_∞ (m/sec)
Oct. 18	60	93×10^6	Early in high-contrast season	25 to 35	120/180	1000
Nov. 25	51	130×10^6	Late in high-contrast season	45 to 55	155/205	100
Dec. 25	53	172×10^6	Early in medium-contrast season	70 to 80	180/230	250

*1973 V_∞ based on Jan 23 maximum launch period of 37 days

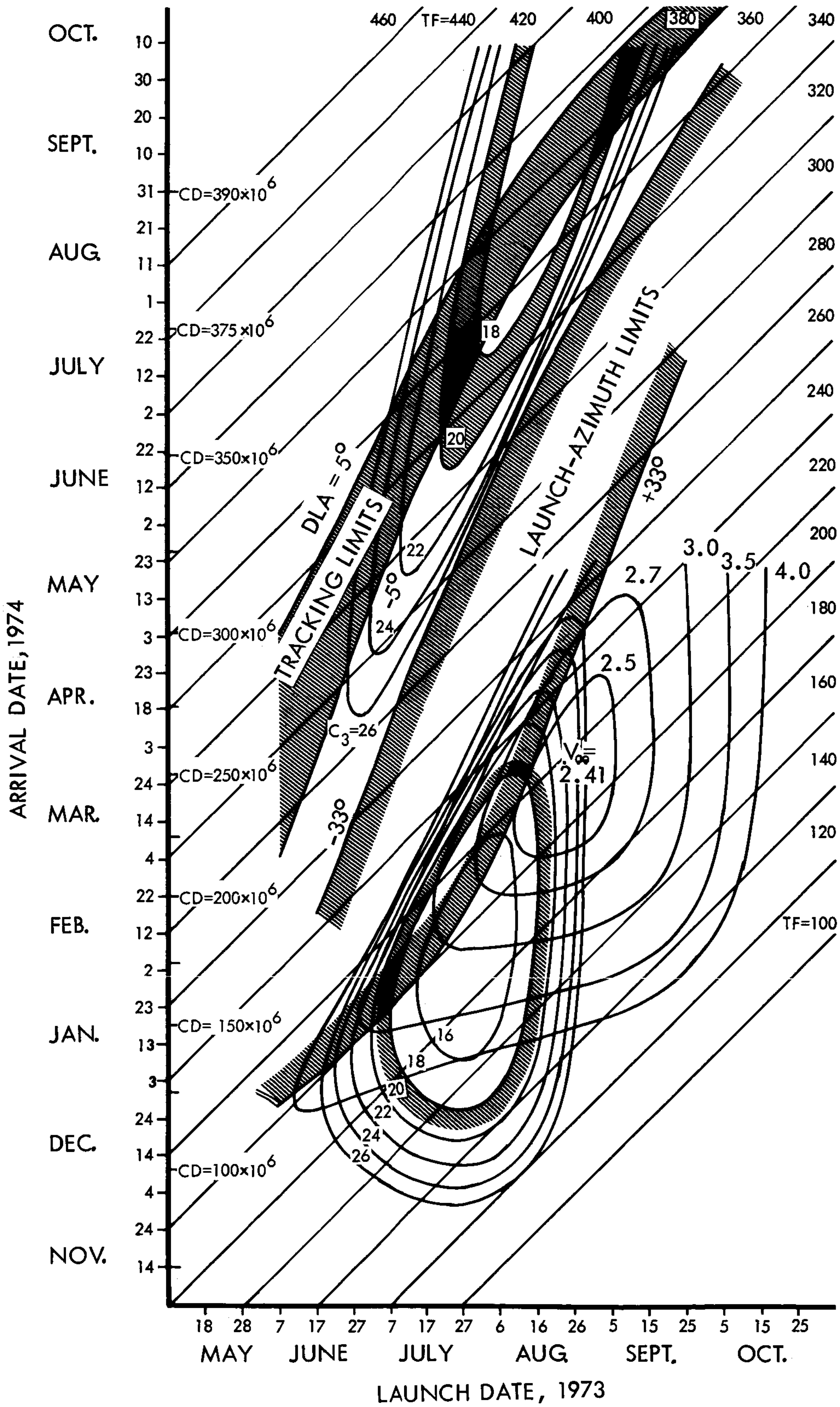


Figure 3.1-3: Trajectory Design Chart, 1973

V_∞ (km/sec)	ARRIVAL DATES	LAUNCH PERIOD (DAYS)	COMMUNICATION DISTANCE AT ARRIVAL (km)	OBSERVATION OF OF DARKENING A ARRIVAL
3.0	NOV 20/DEC 22	49	$128 \times 10^6 / 166 \times 10^6$	LATE IN HIGH CON SEASON TO EARLY IN MEDIUM CONTRAST SEASON
3.5	OCT 31/NOV 15	54	$108 \times 10^6 / 122 \times 10^6$	CENTER OF HIG CONTRAST SEA
4.0	OCT 18/OCT 31	57	$93 \times 10^6 / 107 \times 10^6$	EARLY IN HIGH CONTRAST SEA

WAVE T	OBSERVATION OF WAVE OF DARKENING 60 DAYS AFTER ARRIVAL	ILLUMINATION ANGLE AT PERIAPSIS (DEGREES)	TRA (
TRAST	EARLY IN MEDIUM CONTRAST SEASON TO LATE IN MEDIUM CONTRAST SEASON	50 to 75	17
GH SON	LATE IN HIGH CONTRAST SEASON TO EARLY IN MEDIUM CONTRAST SEASON	35 to 40	14
H SON	CENTER OF HIGH CONTRAST SEASON TO LATE IN HIGH CONTRAST SEASON	30 to 35	13

NSIT TIME DAYS)	LAUNCH PERIOD AVAILABLE IN 1973 WITH CONSTANT V_∞	ALTITUDES AT WHICH ILLUMINATION ANGLE IS FROM 20° TO 50° (km)	PERIAPS ALTITU (km)
'0 to 210	24	5000 to 9500	3800
5 to 185	37	2700 to 4900	2700
5 to 180	29		

COMPARISON OF TYPICAL ORBITS OBTAINED WITH $\triangle V$ = 5

IS DE	NUMBER OF ORBITS/DAY	POSSIBLE NUMBER OF BITS/SEC (TRANSMITTED TYPICALLY)	NUMBER OF BITS/ORBIT ①	RELATIVE RESOLUTION AT PERIAPSIS FOR SAME CAMERA	ORBIT PERIO (HOUR
	2.6	45,000	1.05(x)	0.54	9
	1.3	84,000	(x)	1.0	18

13 (1)

Table 3.1-2: Constant V_∞ Trajectory Sets

700 FPS

) S)	APPROX POSSIBLE BITS/DAY TRANSMITTED	APPROX BITS/DAY ASSUMING SCIENCE PACKAGE PROVIDES A FIXED NUMBER OF BITS/SEC
	1.12(Y)	2.1(Z)
	(Y)	(Z)

① DUE TO DIFFERENCE IN TIME FOR SPACECRAFT TO GO BETWEEN TERMINATORS

5

approach velocity at Mars, V_∞ , of 3.5 kilometers per second. The selection of this constant arrival velocity results in a relatively fixed geometry at Mars encounter throughout the launch period. In addition, a fixed-impulse solid-propellant motor may be used for the orbit-insertion maneuver. The second spacecraft to be launched in the 1971 opportunity will have a trajectory with a nominal approach velocity of 3.25 kilometers per second in order to maintain a minimum of 10 days separation between the two spacecraft. The differential velocity of 250 meters per second (at encounter) between the spacecraft can be used to adjust the arrival geometry to make it nearly the same for the two spacecraft. For this, insertion will be made just prior to the hyperbola periapsis. Alternatively, orbits of different periapsis altitude or period can be selected. Arrival dates will vary from October 31 to November 15 for the first spacecraft and November 10 to November 30 for the second spacecraft.

The resulting set of trajectories for the 1971 opportunity will have: (1) a launch period of 54 days, (2) encounter communication distances ranging from 108 X 10^6 to 140 X 10^6 kilometers, (3) observation of the wave-of-darkening phenomenon in the high-contrast season for the first few days, and (4) illumination angles (measured from the terminator) at periapsis of 35 to 40 degrees.

Launch Trajectory--Launch takes place from AFETR with the Saturn IB/ Centaur launch vehicle. A nominal booster trajectory profile is shown in Figure 3.1-5, which includes the launch vehicle altitude, velocity, flight-path angle, dynamic pressure and load factor as a function of

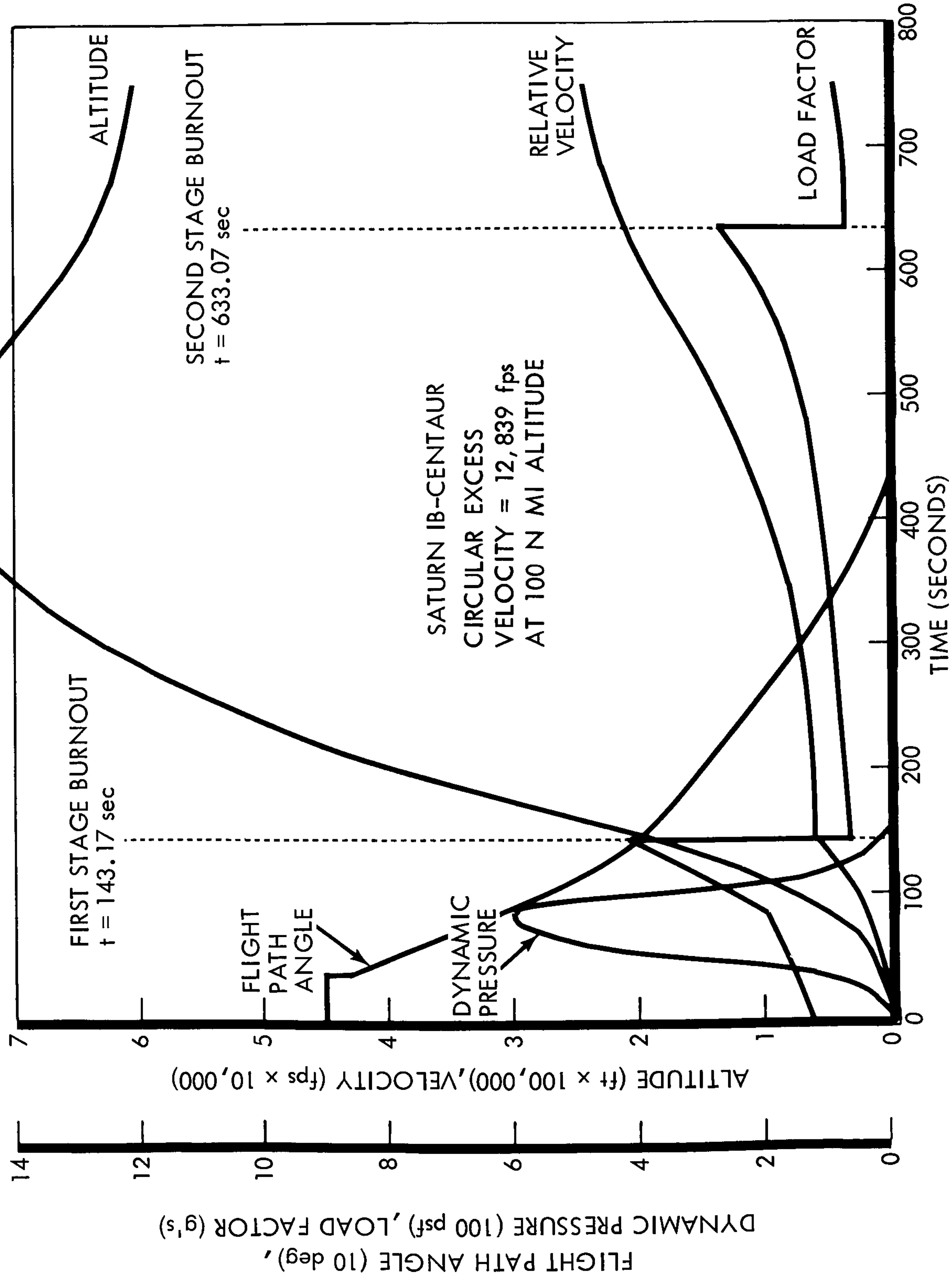


Figure 3.1-5: Saturn IB-Centaur Trajectory Plot

time measured from liftoff. On any given day during the launch period, the right ascension and declination of the geocentric asymptote remain relatively constant. Figure 3.1-6 shows, in red, the range of launch azimuths as a function of time of day for launches on April 30, June 8, and June 23, 1971. These trajectories represent the maximum variations in right ascension and declination (DLA) of the departure asymptote within the 54-day launch period. The allowable launch sector of 71 to 108 degrees for the 1971 mission is indicated on the curve, along with the tracking limitations to absolute values of DLA greater than 5 degrees. A longer than 2-hour launch window is obtained for all these launch dates. Examination of similar curves for the other launch dates indicates a minimum launch window of 2 hours and a maximum launch window of 4.4 hours. The curves in blue on the figure show the various coast times in the parking orbit. Limits on coast time of 2 minutes and 25 minutes are indicated for a typical C_3 (12 km^2/sec^2) by the blue shaded region. The coast times for these particular days vary from 8 to 14 minutes on April 30, from 2 to 18 minutes on June 8, and from 6 to 21 minutes on June 23. The coast times will vary from 2 to 25 minutes over the entire launch period.

Figure 3.1-7 shows the ground track of nominal trajectories for a May 9, 1971 launch date. Four trajectories are shown that represent the typical expected range in the launch azimuth. The locus of injection points is indicated, as well as the rise and set times for various DSN stations. Flight along the outgoing asymptote is essentially established within approximately 15 hours, as indicated at the left portion of the curve.

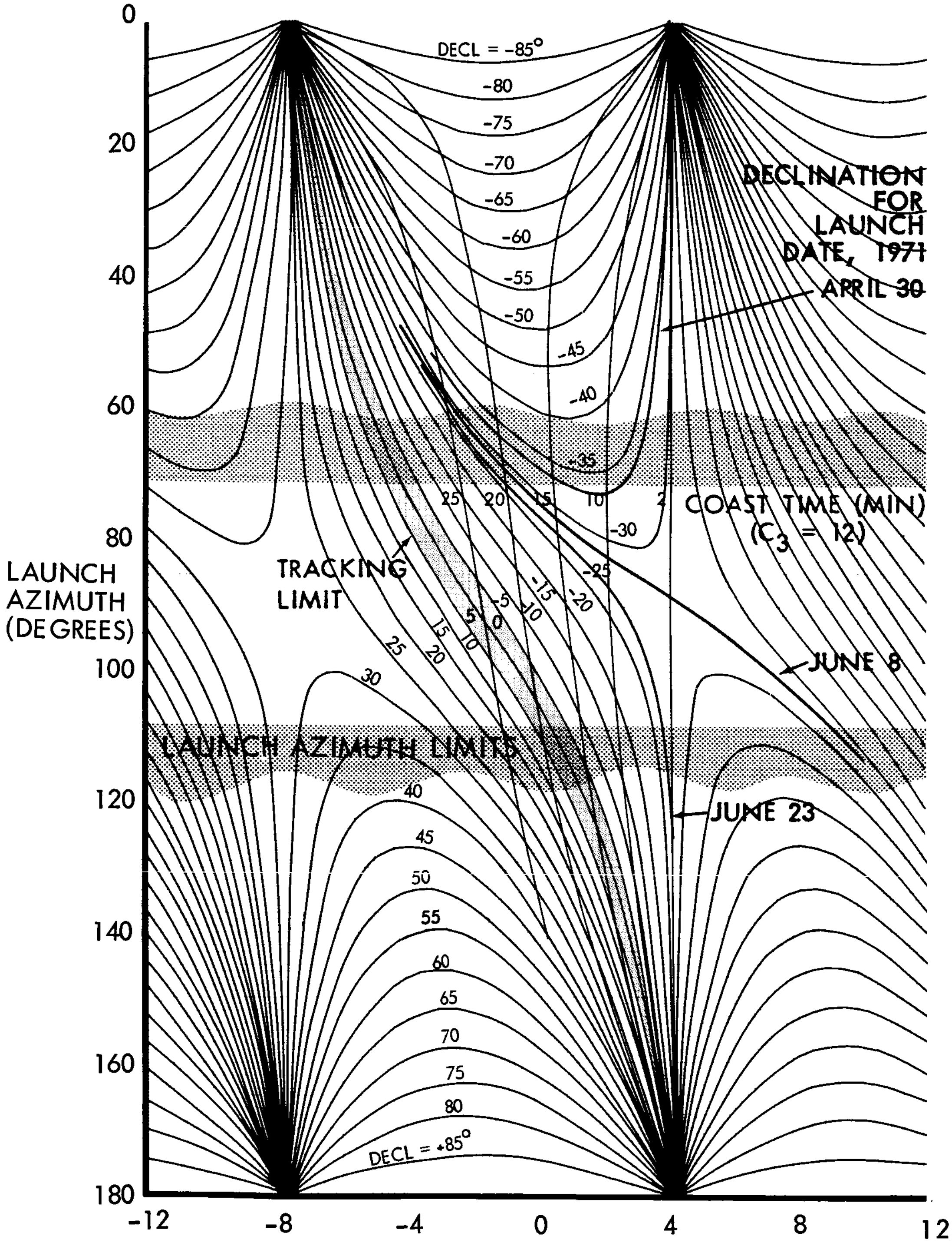
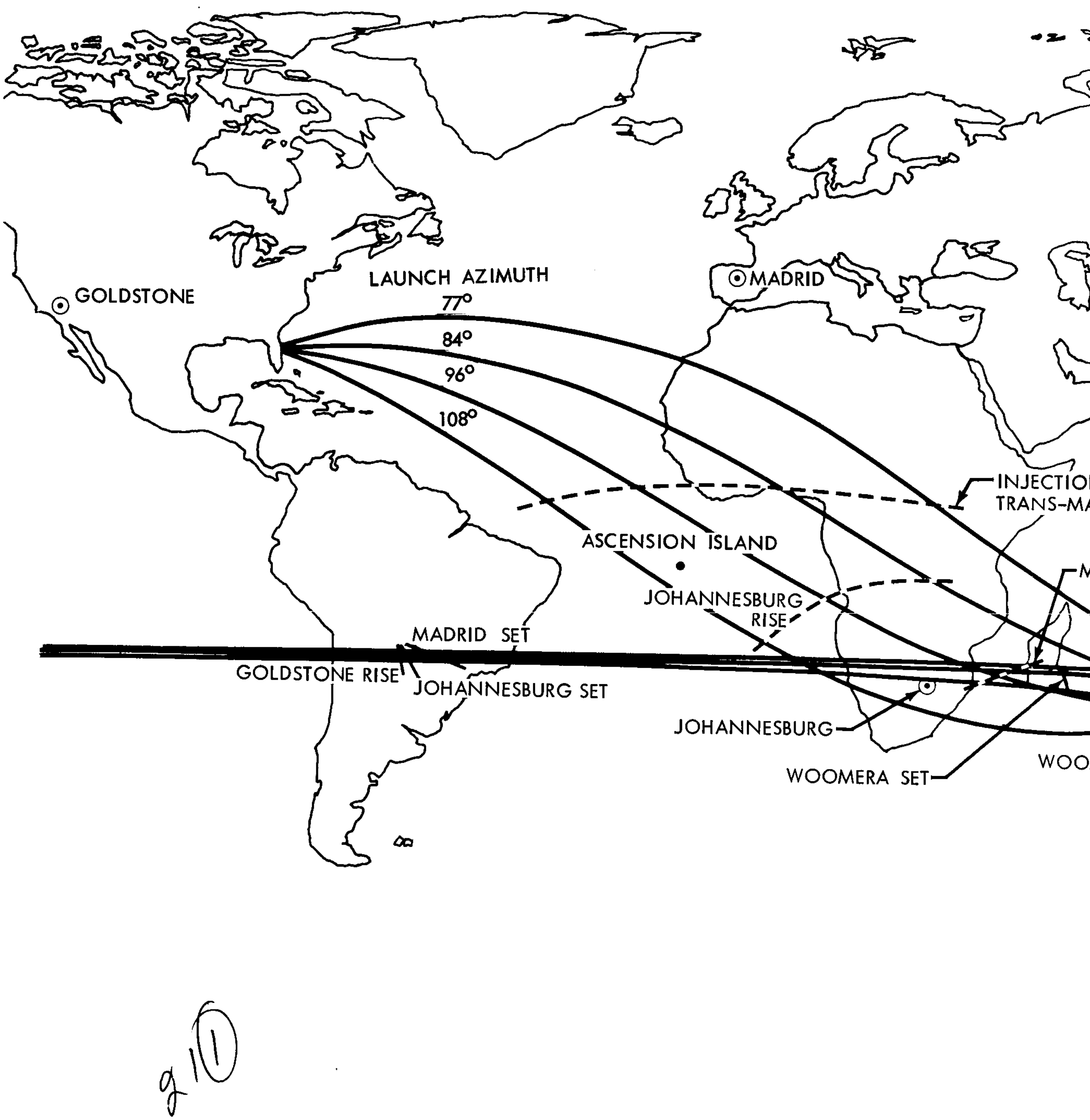


Figure 3.1-6: Launch Azimuth


GOLDSTONE
LAUNCH AZIMUTH
77°
84°
96°
108°
MADRID
INJECTIO
TRANS-MA
ASCENSION ISLAND
JOHANNESBURG
RISE
MADRID SET
GOLDSTONE RISE
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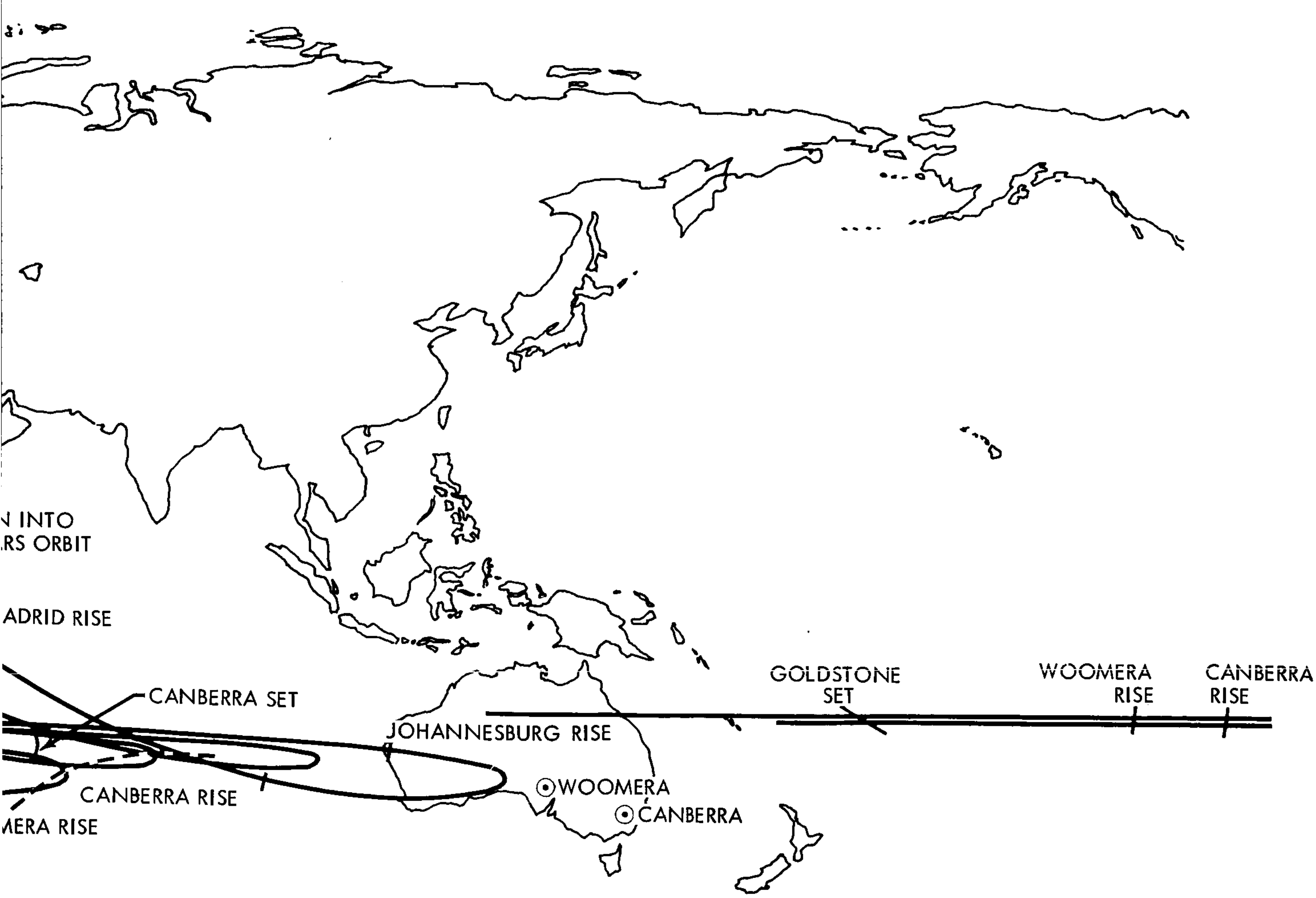


Figure 3.1-7: Ground Tracks for Trajectories Launched May 9, 1971 Arriving Dec. 25, 1971

2

Near-Earth Trajectory--The near-Earth trajectory phase begins at departure from the Earth parking orbit with injection into the Mars transit trajectory.

Following injection into the Mars transit trajectory and separation from the Centaur stage, the spacecraft moves along on Earth escape trajectory until the Sun's gravitational influence becomes dominant over that of the Earth. The typical variation in altitude with time during the near-Earth trajectory phase is indicated in Figure 3.1-8 for May 9 launches. The altitude is seen to increase to 400,000 kilometers within 30 hours.

Figure 3.1-9 contains the doppler parameters, that is, range, range rate, and acceleration along the line of sight for the Johannesburg deep-space station. Hour angle and declination are presented in Figures 3.1-10 and 3.1-11. The rise and set times are indicated for elevation angles of 10 degrees above the horizon. The time reference on these charts is from the time of launch. Similar data has been calculated for the Woomera, Goldstone, Madrid, and Canberra DSN stations.

Heliocentric Flight--During the heliocentric phase of flight, the Sun is the dominant body and the planetary vehicle moves in a near-elliptical trajectory. Figure 3.1-12 depicts the heliocentric phase of a typical flight. The orbits of Earth, Mars and the transit trajectory are indicated with the corresponding distances. The pertinent orbital parameters for Earth and Mars are shown on the sketch. Figure 3.1-13 shows the launch dates and arrival times for a V_∞ = 3.5 kilometers per second and a V_∞ = 3.25 kilometers per second on a trajectory design chart. The

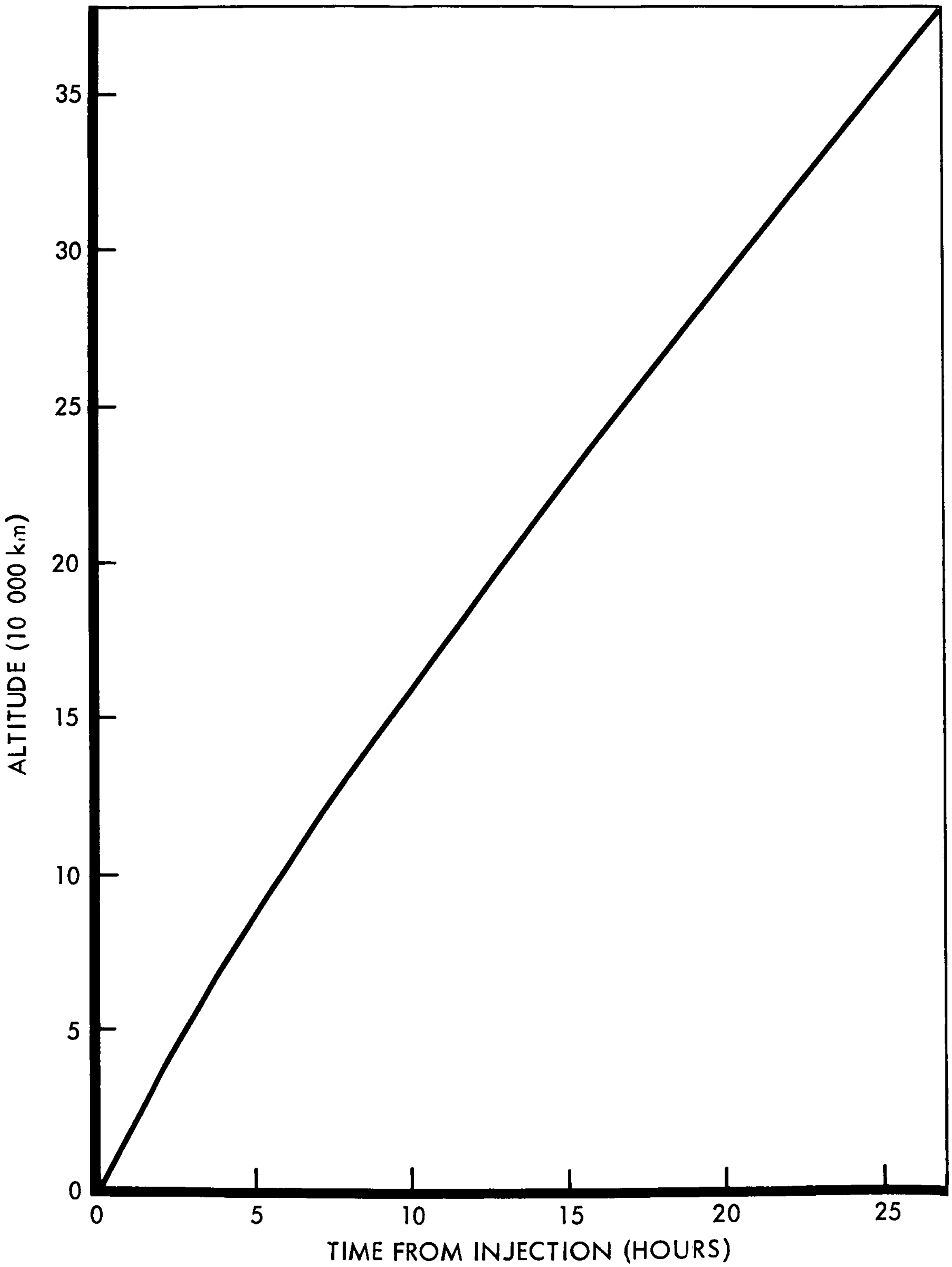


Figure 3.1-8: Near-Earth Altitude (C_3 = 9.82 km^2/sec^2)

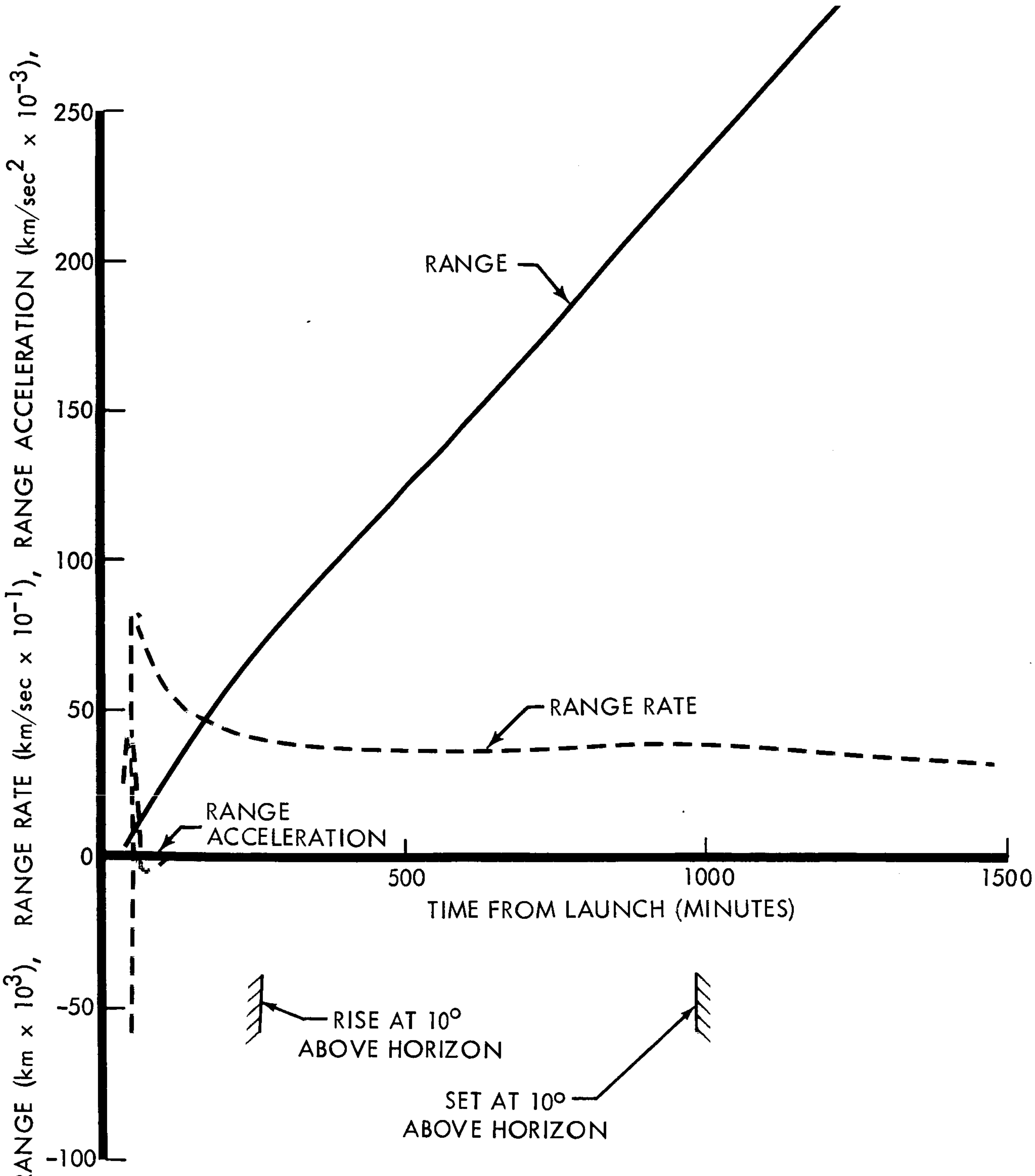


Figure 3.1-9: Johannesburg Doppler Data Range

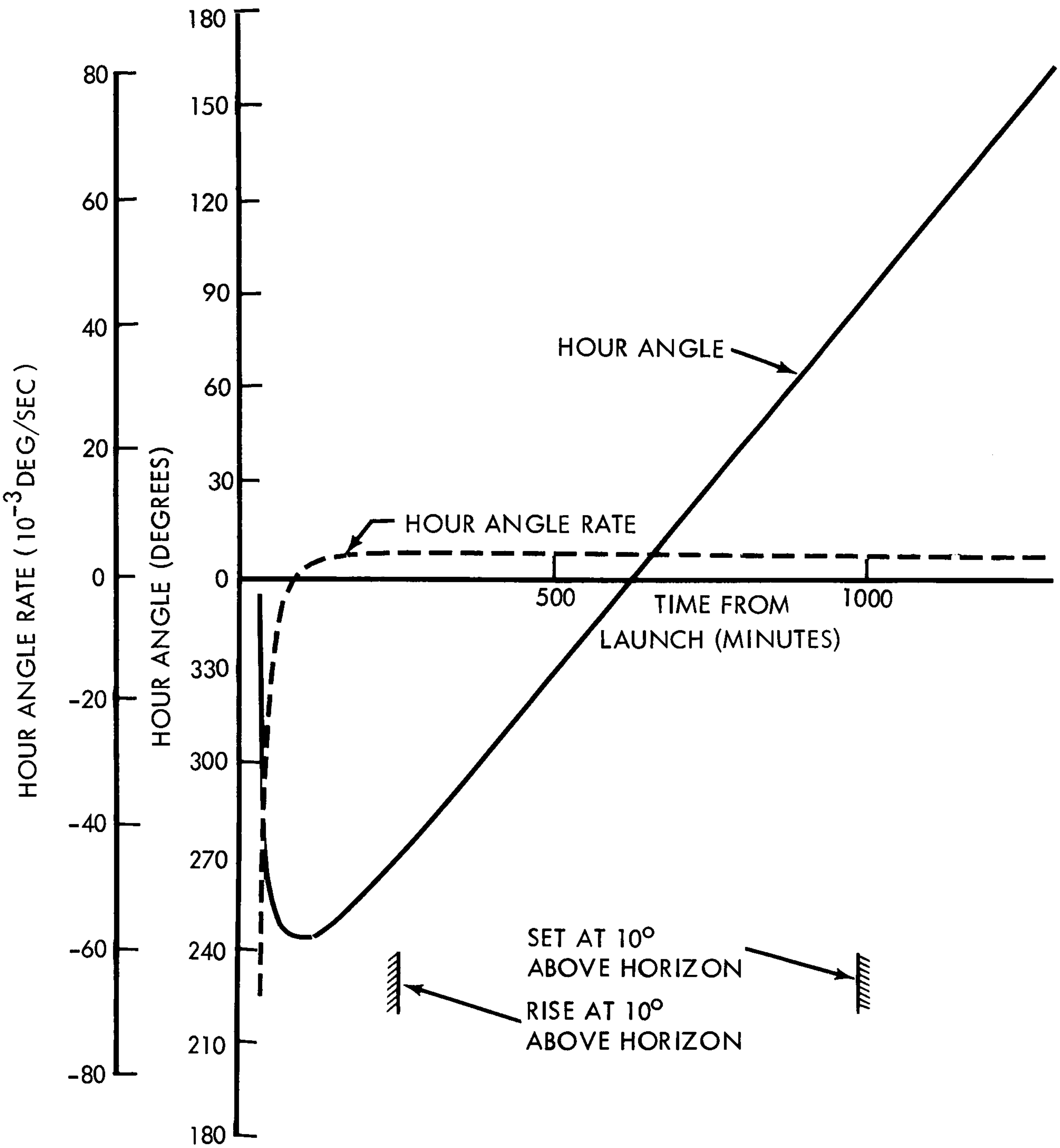


Figure 3. 1-10: Johannesburg Tracking Data; Hour Angle

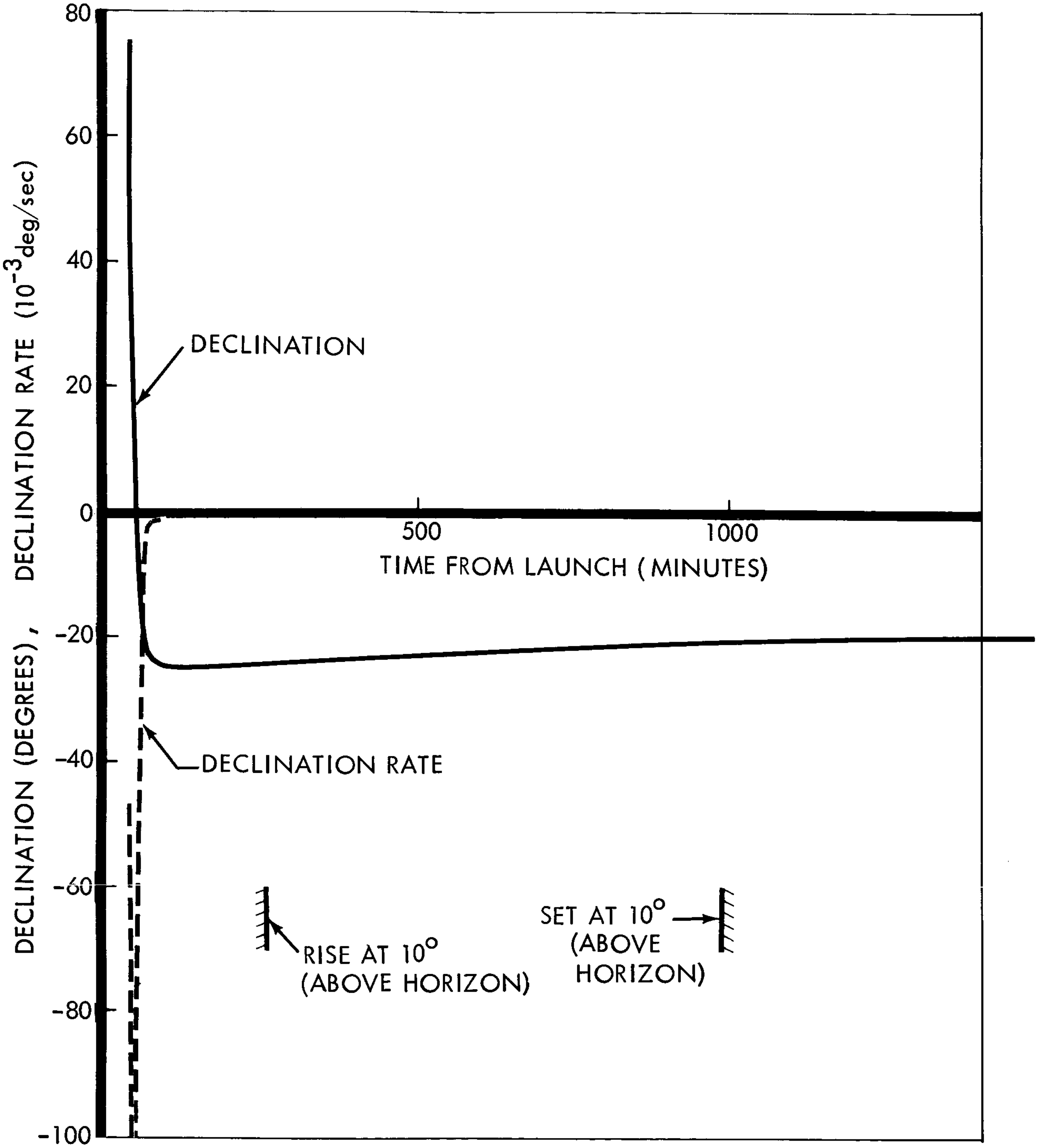


Figure 3.1-11 : Johannesburg Tracking Data; Declination

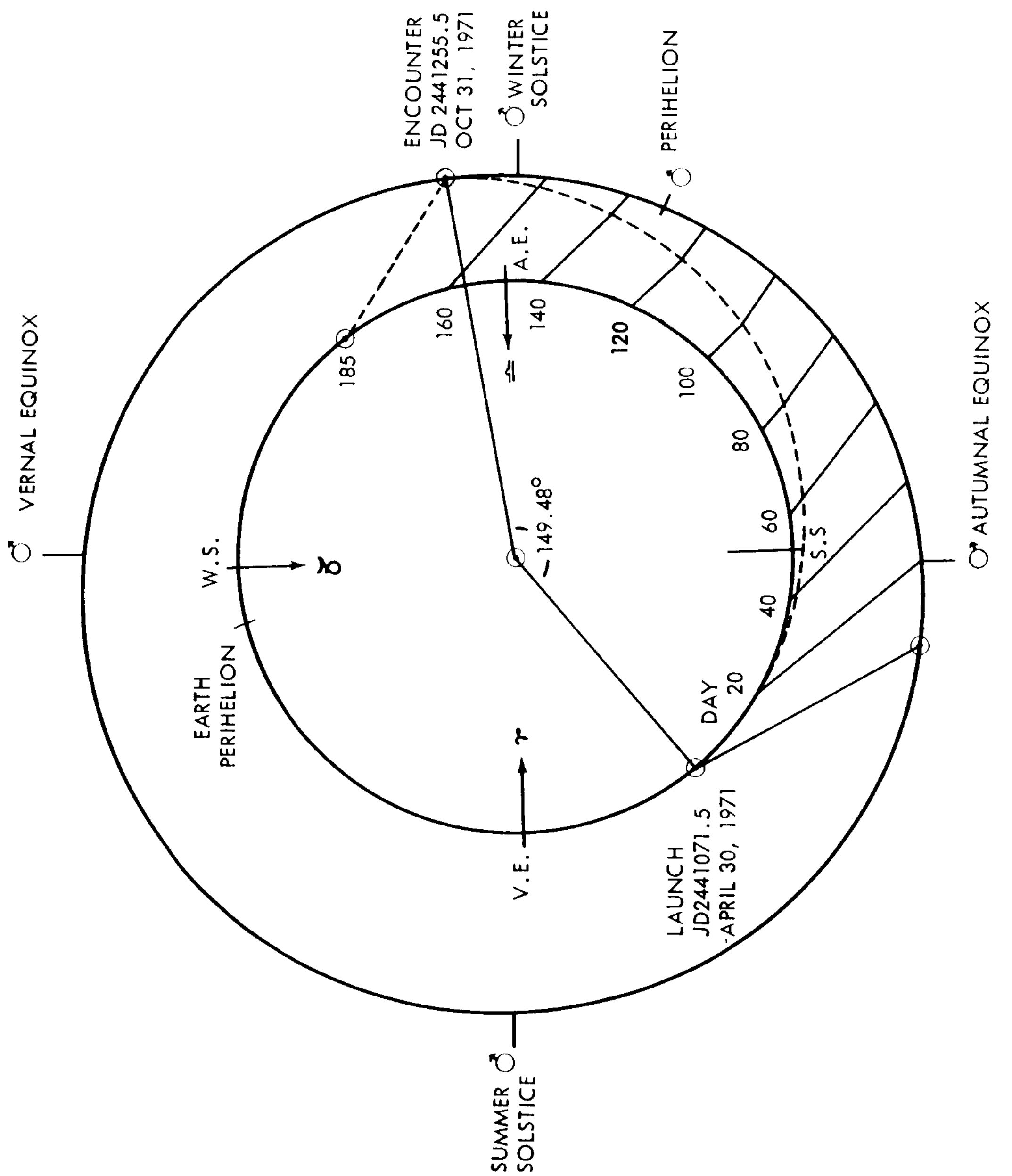


Figure 3.1-12: Typical Transit Trajectory

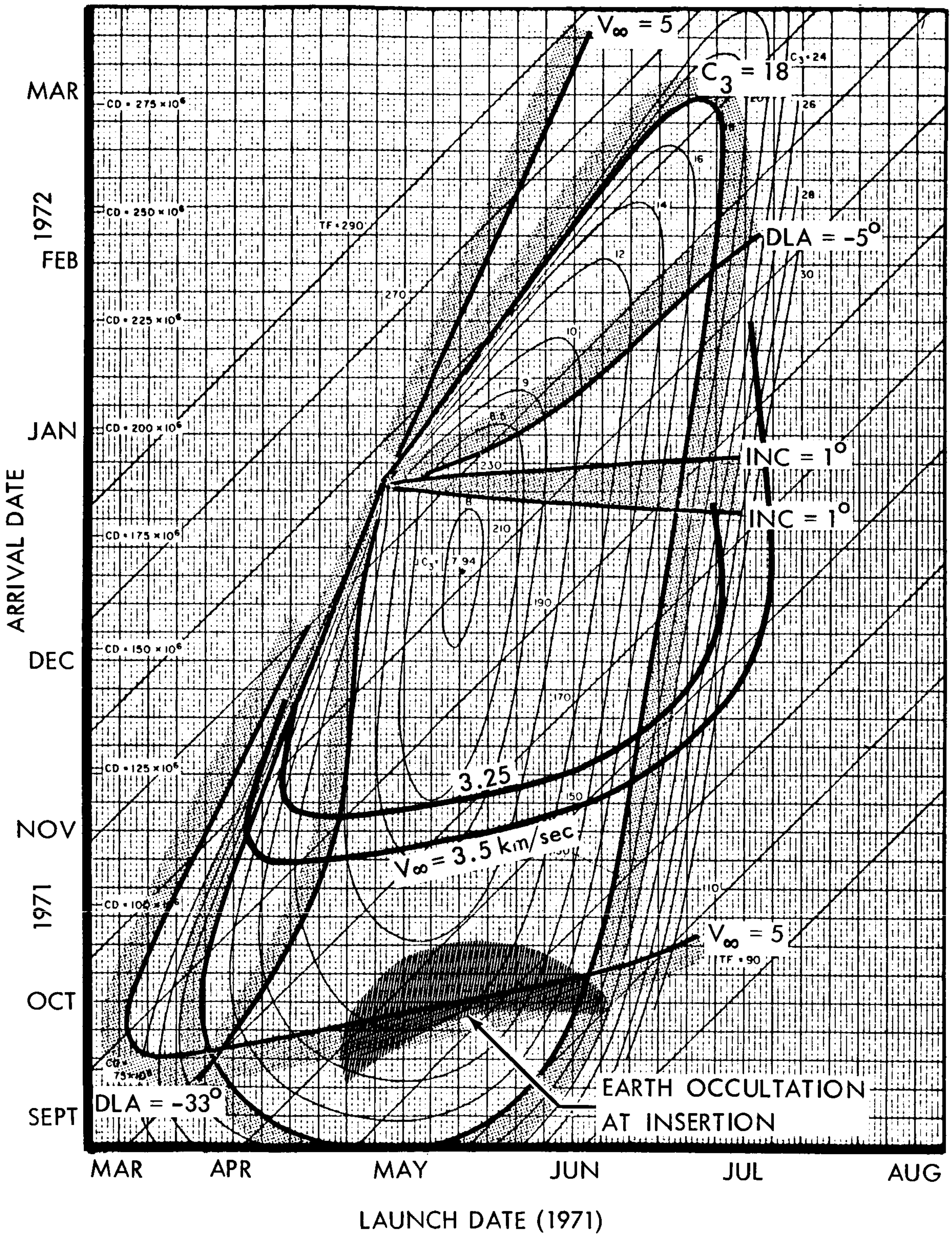


Figure 3.1-13: Trajectory Design Chart for 10-Day Separation

figure shows the C_3 requirements as a function of the launch date and arrival date. The declination of the geocentric asymptote is shown in Figure 3.1-14 over the range of launch dates, April 30 to June 23, 1971. This determines the daily launch window for a given launch azimuth sector. Figure 3.1-15 shows the C_3 that is necessary to depart the Earth and establish the Mars transit trajectory for the various launch and arrival dates under consideration. The launch period is indicated by the boundaries of the curve. The maximum C_3 allowable is 18 km^2/sec^2, which is the limit for the latest launch date.

The Earth cone and clock angles for the example trajectory set are shown in Figures 3.1-16 and 3.1-17. There is a maximum variation in the cone angle of 85 degrees, which occurs the first few days after departure. The clock angles for each trajectory have approximately the same limits. The communication distance as a function of time from launch is shown in Figure 3.1-18. The communication distance at arrival will vary from 106×10^6 to 122×10^6 kilometers.

3.1.3 Capsule Separation

The selection of the capsule separation mode and the time of separation depends on: (1) maintenance of planetary quarantine, (2) minimum dispersions in the landing site, (3) separation velocity requirements, and (4) communication from the capsule to the spacecraft until landing.

The aiming point for the spacecraft was selected for maintenance of planetary quarantine. Any small velocity impulse imparted to the spacecraft, in the direction of Mars, will increase the probability of impacting

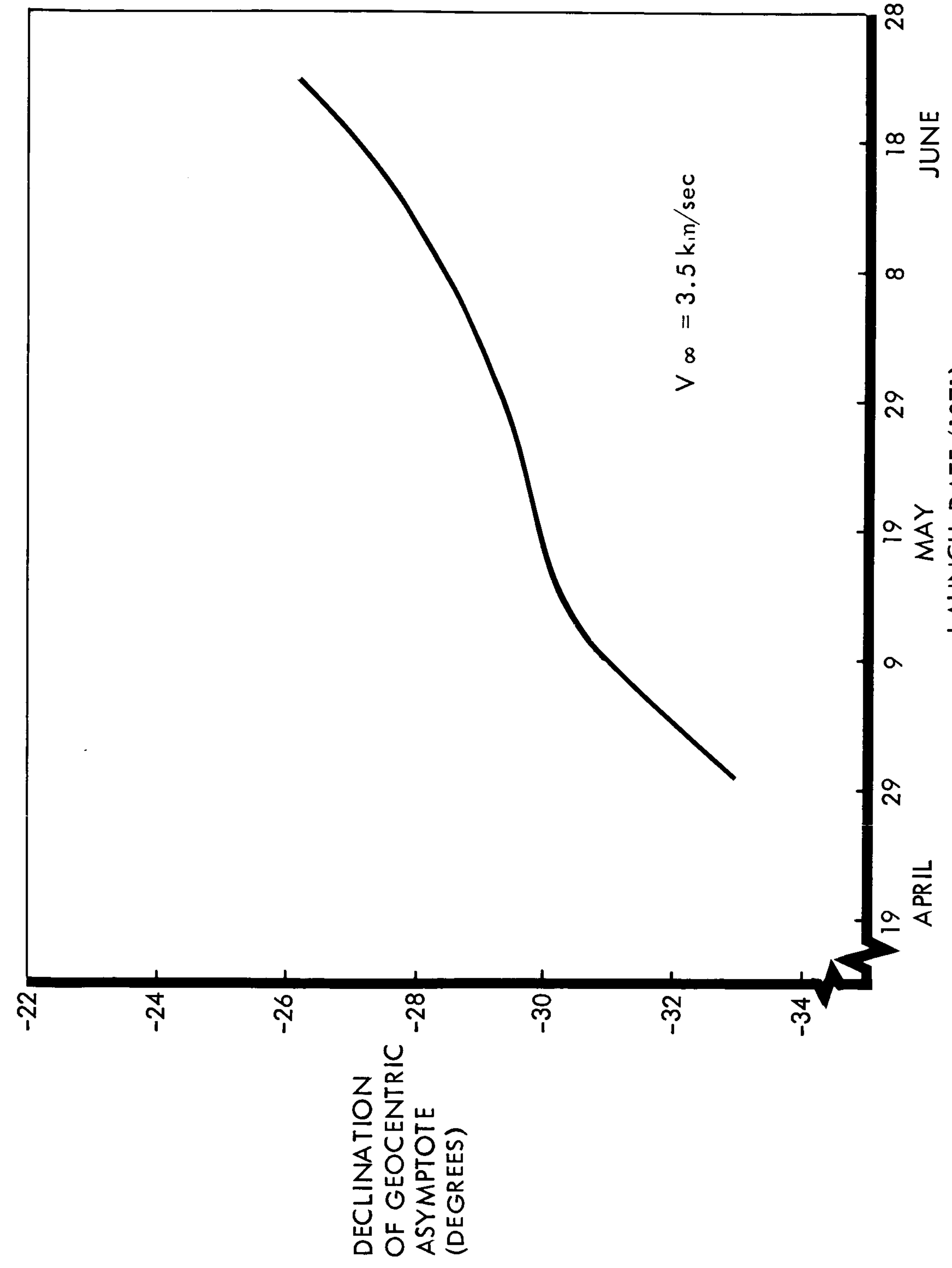


Figure 3.1-14: Departure Declination

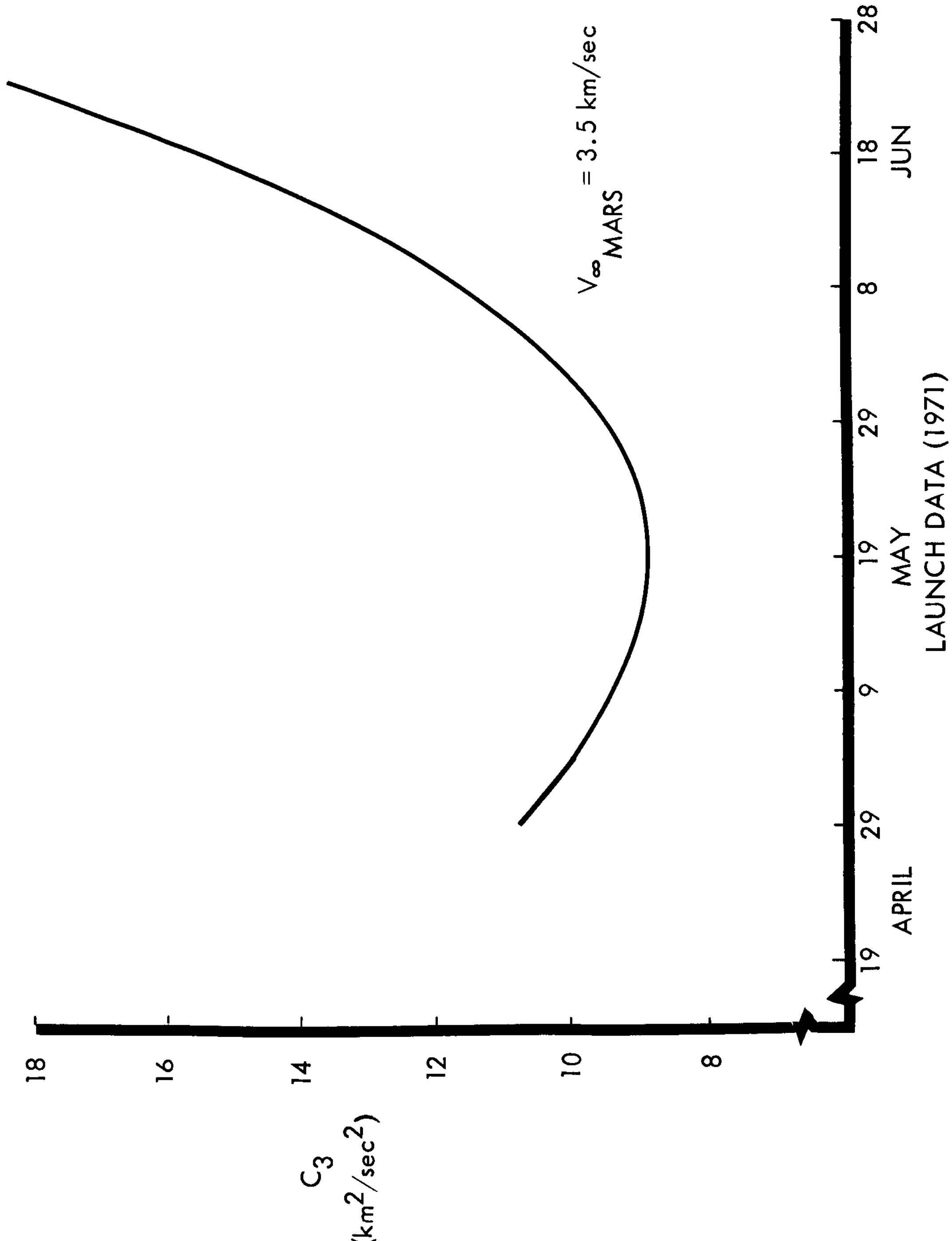


Figure 3.1-15: Launch Energy

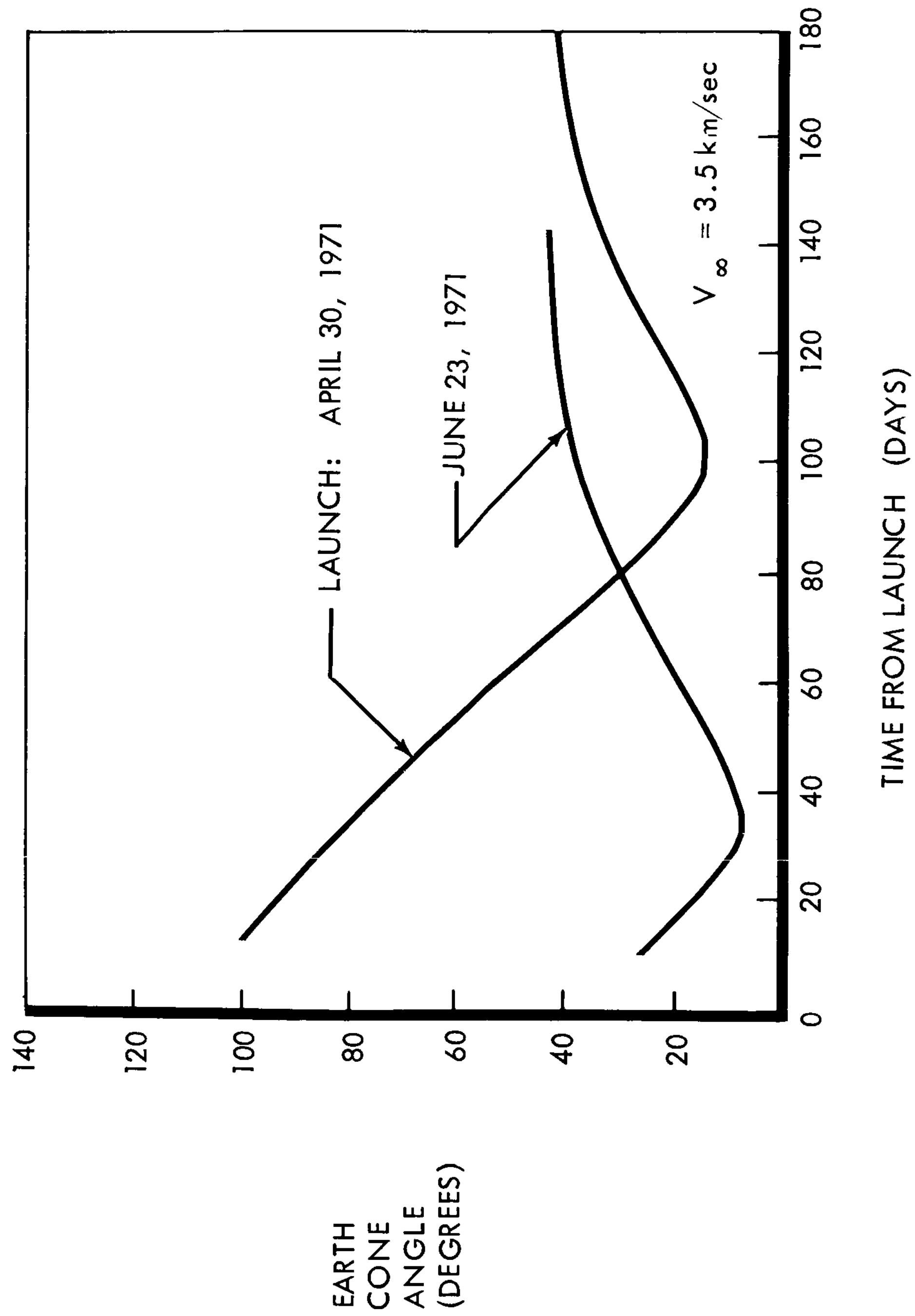


Figure 3.1-16: Earth Cone Angles

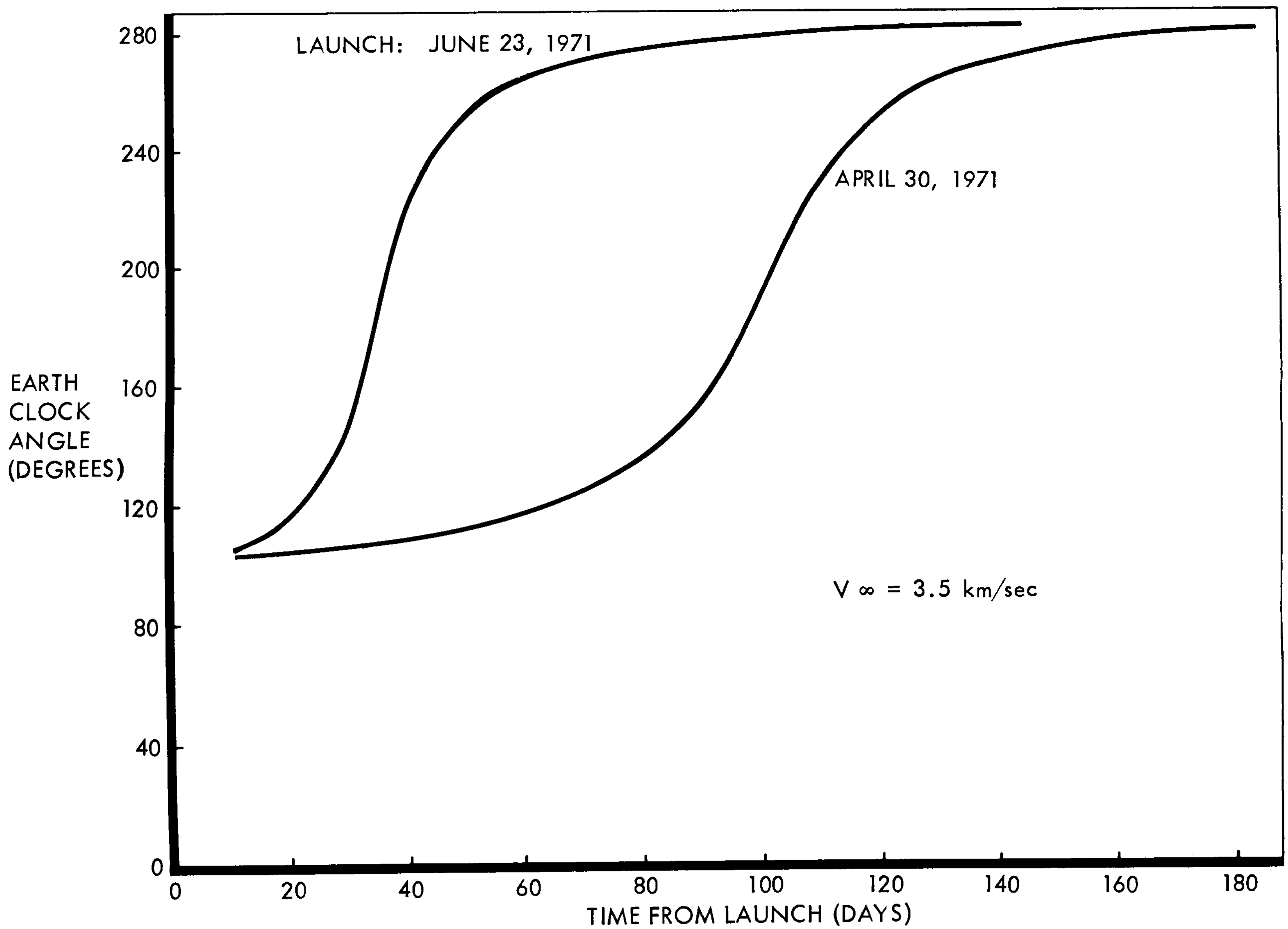


Figure 3.1-17: Earth Clock Angle, Cruise Phase — Trajectories with Arrival at 3.5 km/sec

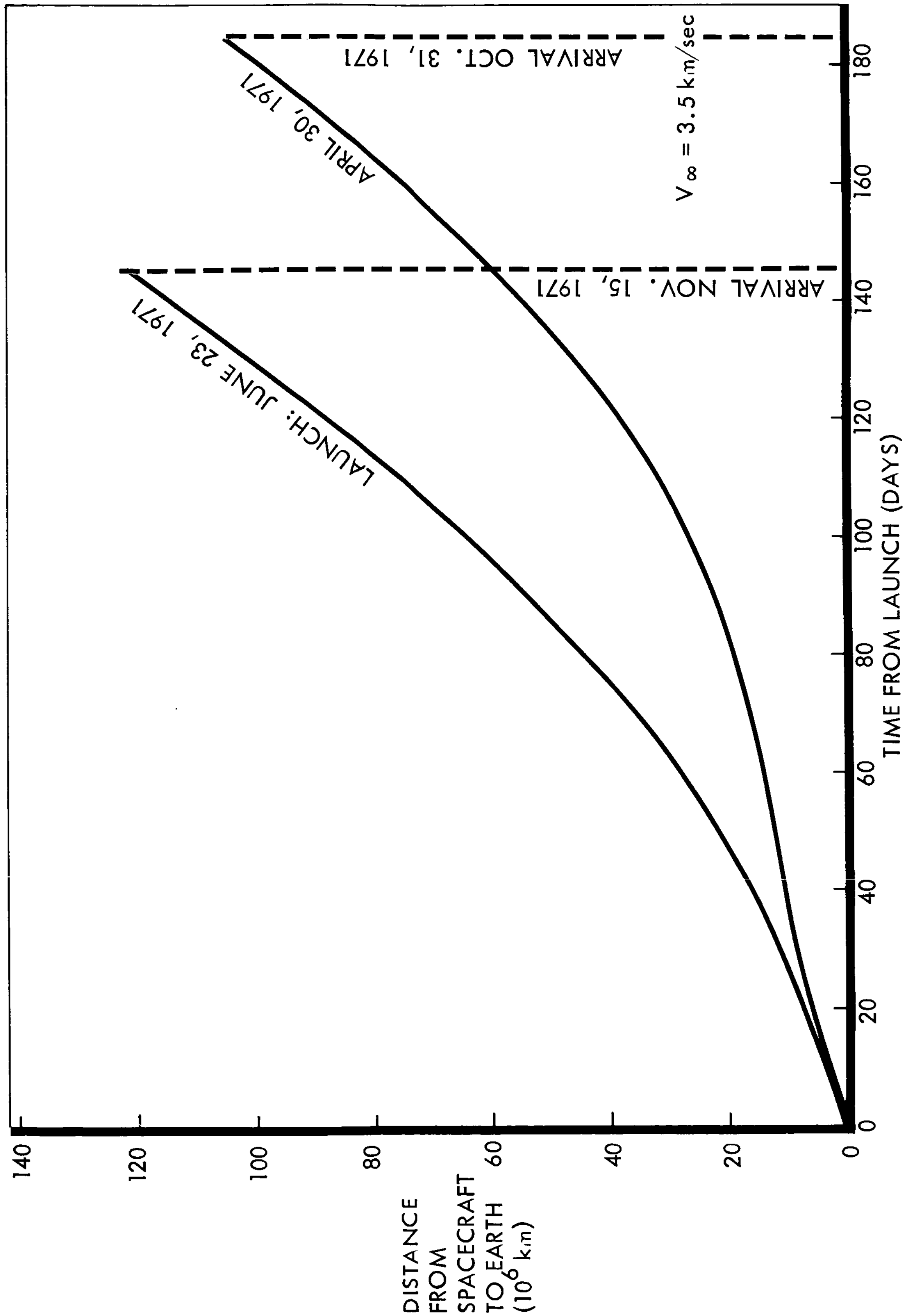


Figure 3.1-18: Communication Distance

the planet over a 50-year period. As a result, none of the separation maneuvers considered will impart a significant velocity impulse to the spacecraft. Figure 3.1-19 shows the expected dispersions in the landing-site location for various times from separation to entry. A slight minimum in the size of the dispersion ellipse occurs for separation 3 days from entry. However, this minimum is so small that it will not significantly influence the selection of the separation time.

Within the constraint for maintaining communication between the capsule and spacecraft through landing, capsule deflection ΔV is smaller for earlier separation times as shown in Figure 3.1-20. For example, separation of the capsule and spacecraft 5 days away from Mars will require a ΔV of 35 meters per second for a communication time of 2 hours after entry.

3.1.4 Mars Orbits

3.1.4.1 Mars Orbit Selection Criteria

Only those Mars orbits that satisfy the requirements imposed by the aiming point selection in Section 2.4 will be used. This region is shown in the basic aiming point chart of Figure 3.1-21. This chart is a composite that shows the loci of occultation regions at Mars for a V_∞ of 3.5 kilometers per second during the launch period. There is an acceptable region (from $\theta = 5$ degrees to $\theta = 145$ degrees) which is free from pre-insertion occultation of Sun, Canopus, and Earth. Mars orbits that satisfy the aim-point constraints have been evaluated on the basis of the following criteria: (1) maintenance of planetary quarantine,

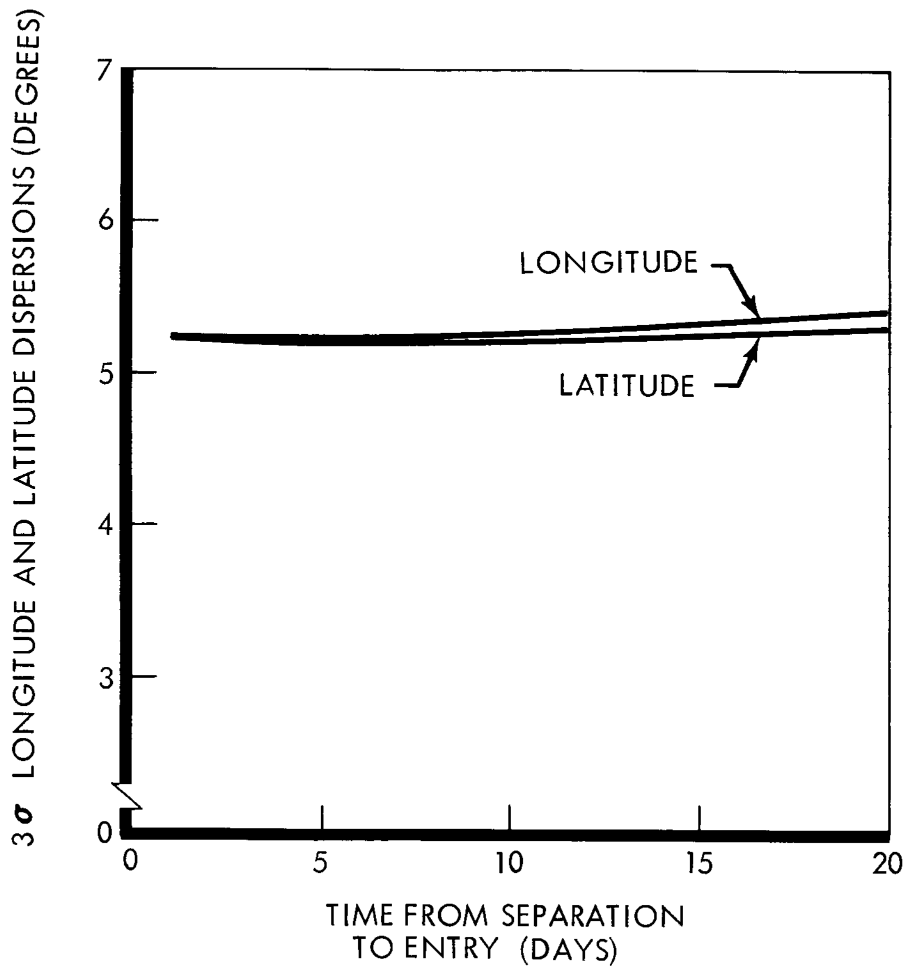


Figure 3.1-19: Typical Capsule Landing Dispersions

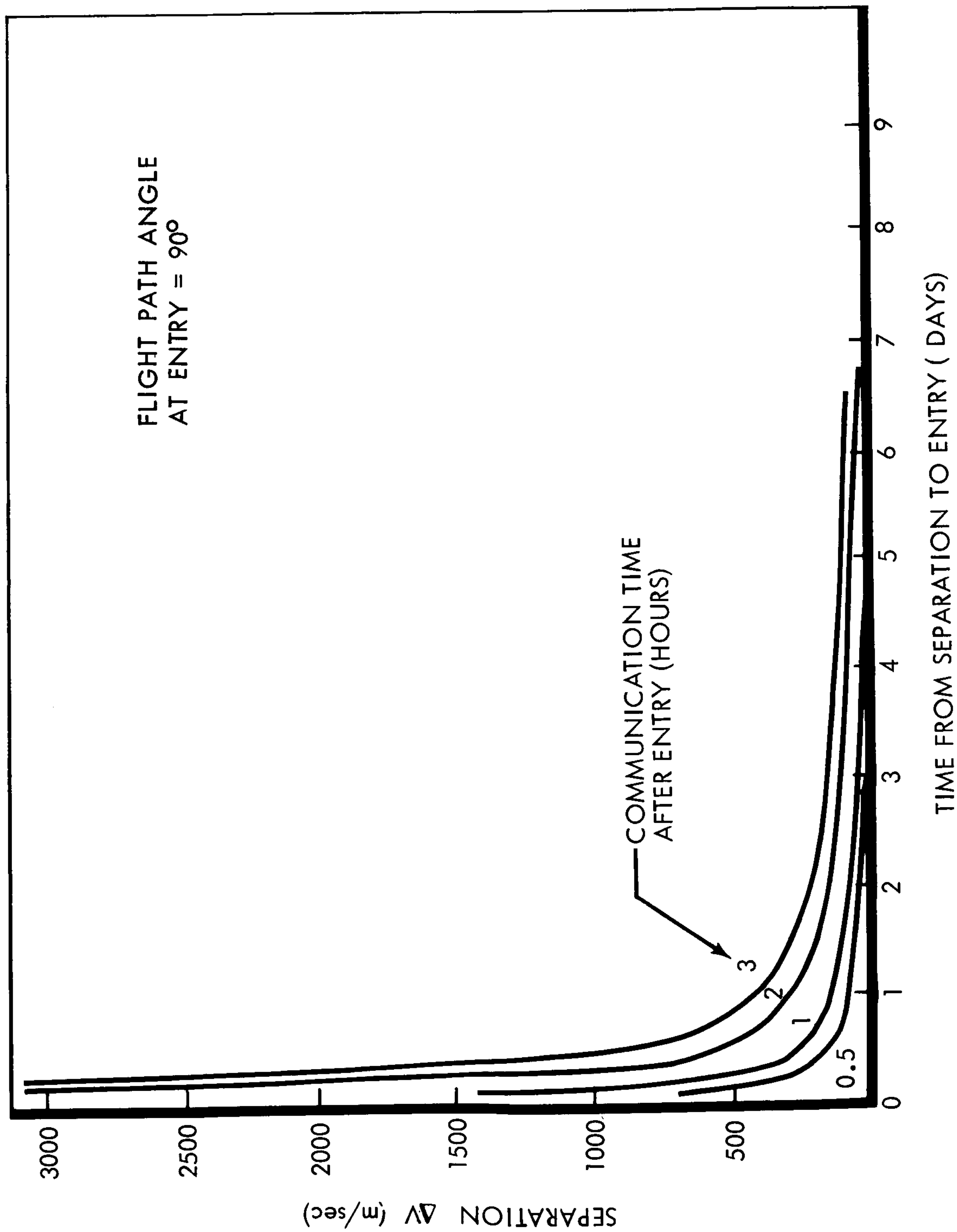


Figure 3.1-20: Capsule/Spacecraft Separation Distance

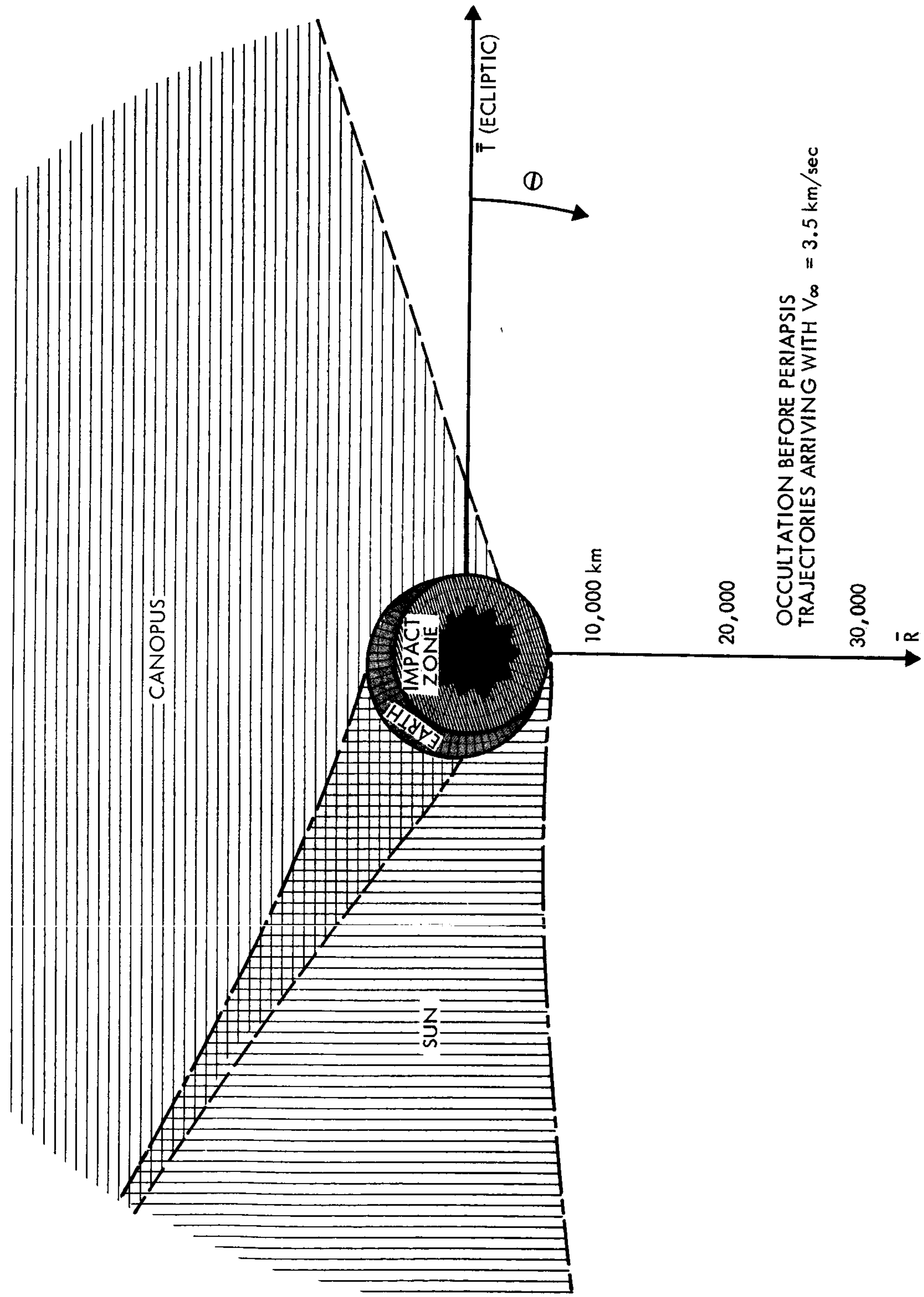


Figure 3.1-21: Aiming Point Constraints

(2) latitude coverage, (3) illumination, (4) communication to Earth, (5) freedom from solar and Canopus occultation, (6) periapsis altitude and orbit period, (7) insertion velocity requirements and (8) compatibility with the 1973 mission. The orbit size (periapsis altitude and period) is considered first. The orbit size is selected on the basis of planetary quarantine, telemetry time per orbit, absence of long solar occultation, satisfactory ground coverage and insertion velocity requirements. The orbit inclination is selected on the basis of latitude coverage, illumination, and occultation.

3.1.4.2 Orbit Size

The orbit size is strongly affected by the planetary quarantine requirement. Figure 3.1-22 shows a curve of periapsis altitude, apoapsis altitude and ballistic coefficient (M/C_DA) for a 50-year lifetime. For the preferred configuration discussed in Section 3.10, tumbling about a single axis results in a net frontal area of 146 square feet. This gives the spacecraft an M/C_DA of approximately 0.2 slugs per square foot. For M/C_DA = 0.2 slugs per square foot and example orbit periods of 8 to 24 hours, the allowable range of minimum periapsis altitudes is 1750 to 3600 kilometers. Figure 3.1-22 is based on a nonrotating atmosphere (NASA-JPL upper limit). The effect of a rotating atmosphere in the presence of solar perturbations has been investigated. The actual orbit lifetime will be slightly longer than indicated in Figure 3.1-22 due to the atmosphere rotation. Figure 3.1-23 shows the allowable range of periapsis altitudes and orbit periods for the spacecraft's orbit-insertion propulsion-system capability of 5700 feet per second. The minimum orbit periapsis altitude is determined by the 50-year orbit lifetime requirement. This lower limit for periapsis is shown for various orbit periods.

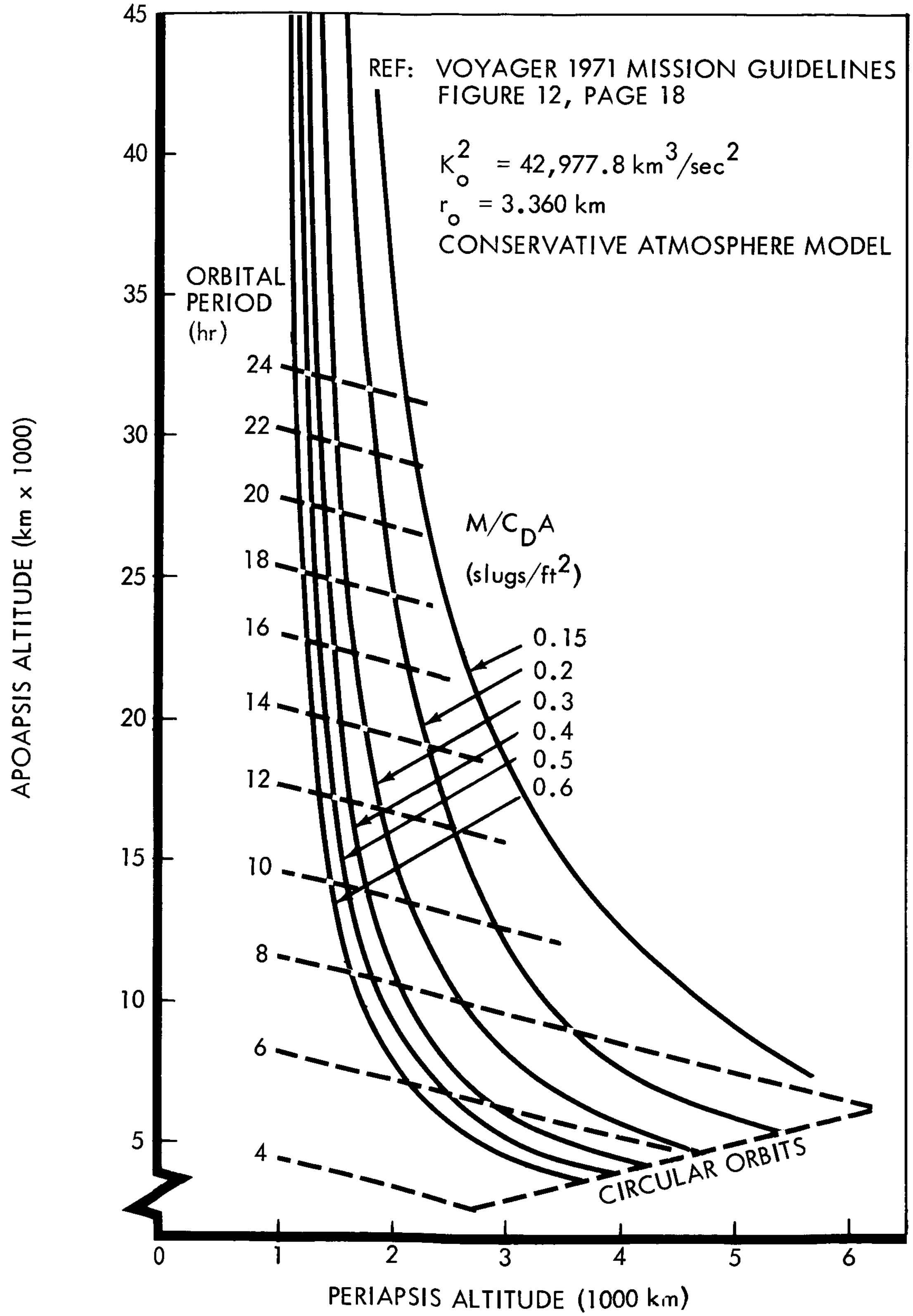


Figure 3.1-22: Mars Orbits for 50-Year Lifetime

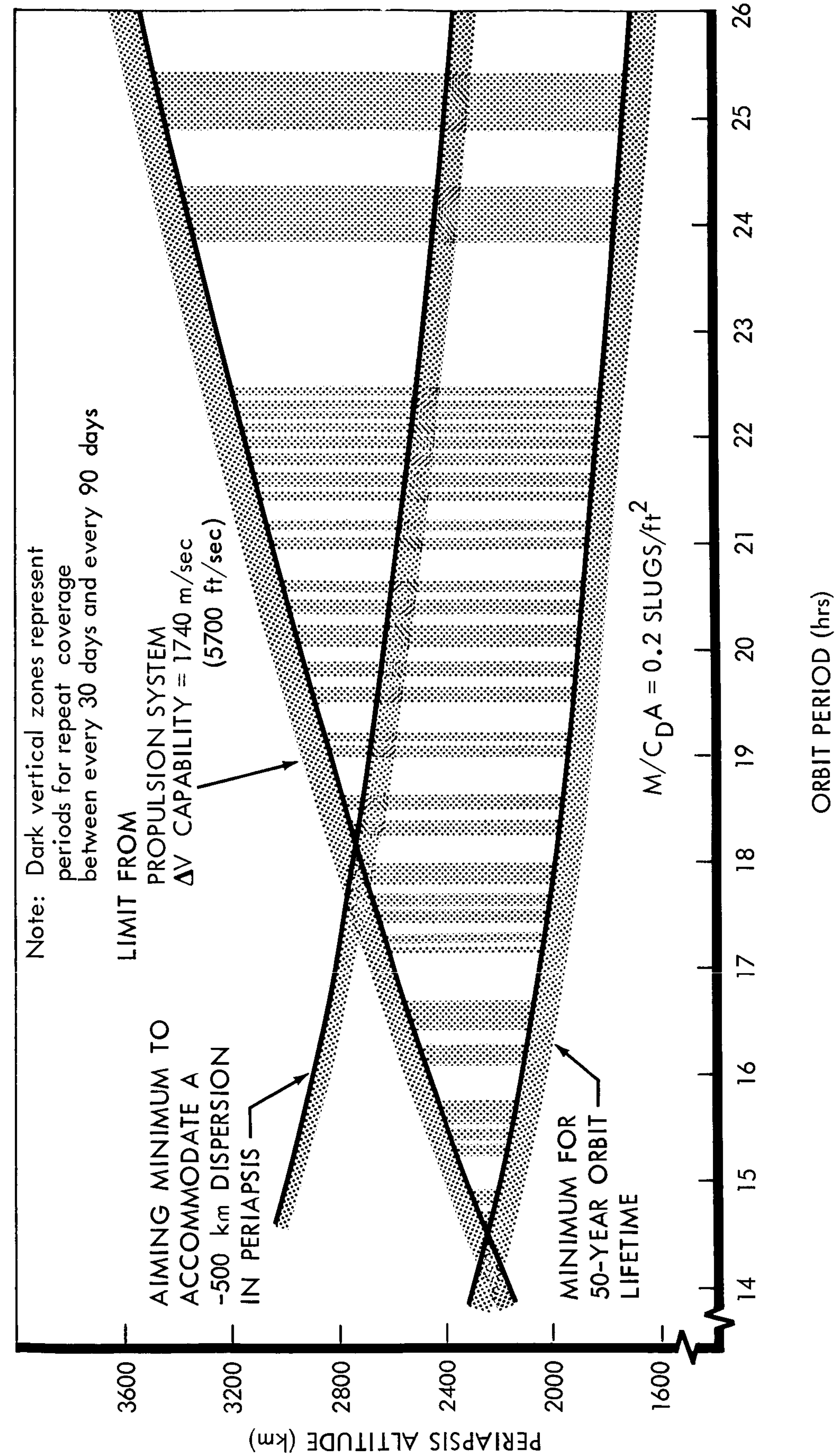


Figure 3.1-23: Orbit Size Limits

Due to guidance dispersions, the mission aim point must be for a higher periapsis, as discussed in Section 2.4. As an example, Figure 3.1-23 also shows the aimed periapsis required to accommodate a -500 kilometer dispersion in periapsis, but still obtain a safe orbit using 5700 feet-per-second. Insertion at periapsis from an approach velocity (V_∞) of 3.5 kilometers per second is assumed.

The upper limit on orbit size depends on the propulsion system capability. The example limits for 5700 feet per second with a V_∞ = 3.5 kilometers per second are also shown. Orbits that satisfy the propulsion limits and the quarantine constraints fall in the V-shaped region in the upper right portion of the plot. The dark vertical bands indicate orbit periods, which will return to the same location every one to 3 months.

3.1.4.3 Orbit Inclination

Figure 3.1-24 shows Sun, Canopus, and Earth occulation regions for various orbit inclinations, as a function of time from periapsis. This composite curve shows approach conditions for an April 30 launch date and a June 8 launch date. These are the dates which are at the limits of the possible approach vector positions for a constant V_∞ = 3.5 kilometers per second trajectory set. The regions of inclinations free from pre-insertion occulation are also shown by the shading adjacent to the graph's ordinate. The desired limits of latitude coverage (10°N to 40°S) prefer an inclination greater than 40 degrees. This chart is for the first orbit after insertion. Figures 3.1-25 and 3.1-26 show occultation regions during the 40th and 80th orbits. In these preliminary design figures the Canopus occultation region assumes sensor failure if

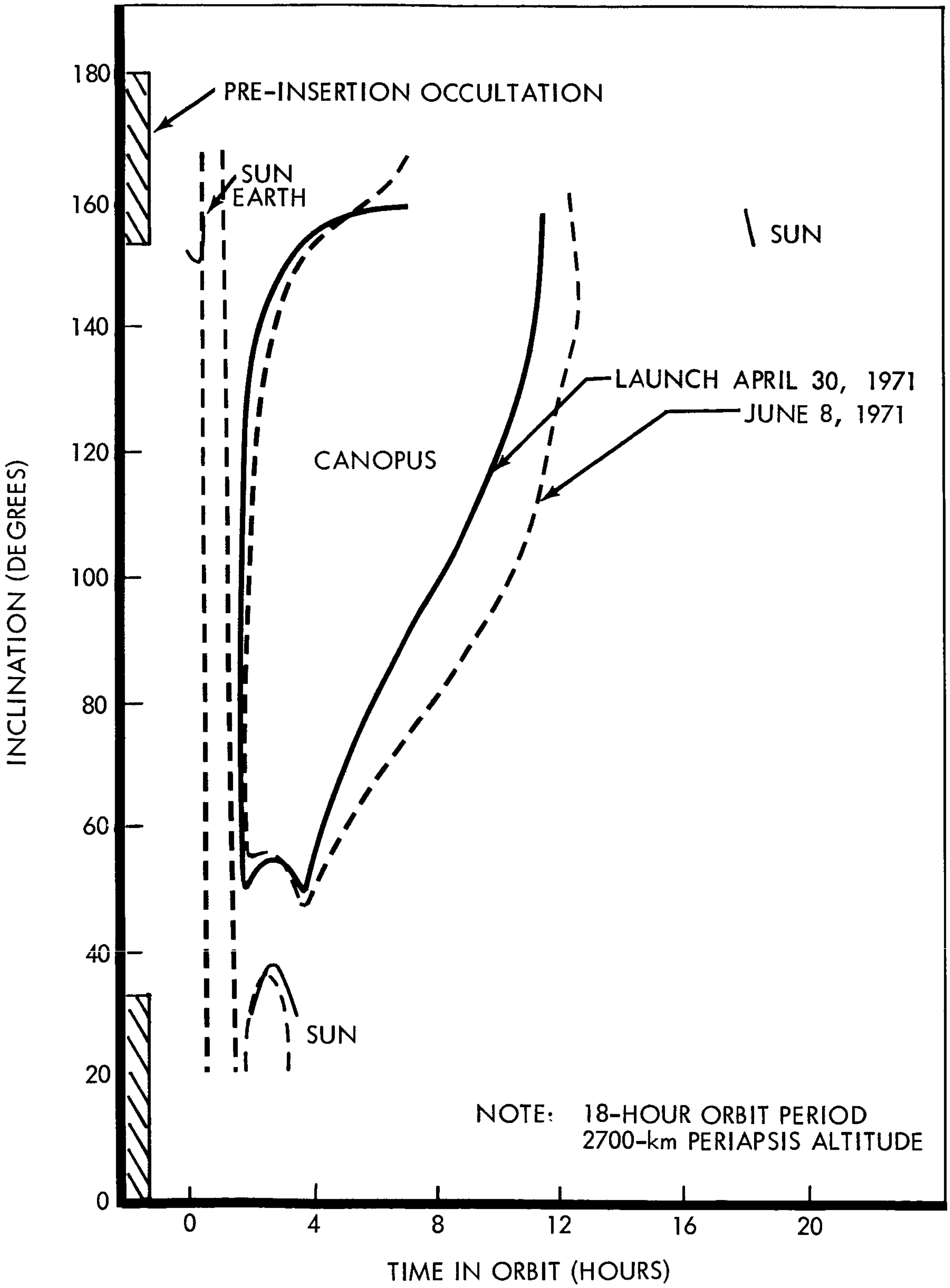


Figure 3.1-24: Occultation in Orbit — First Orbit (First Day)

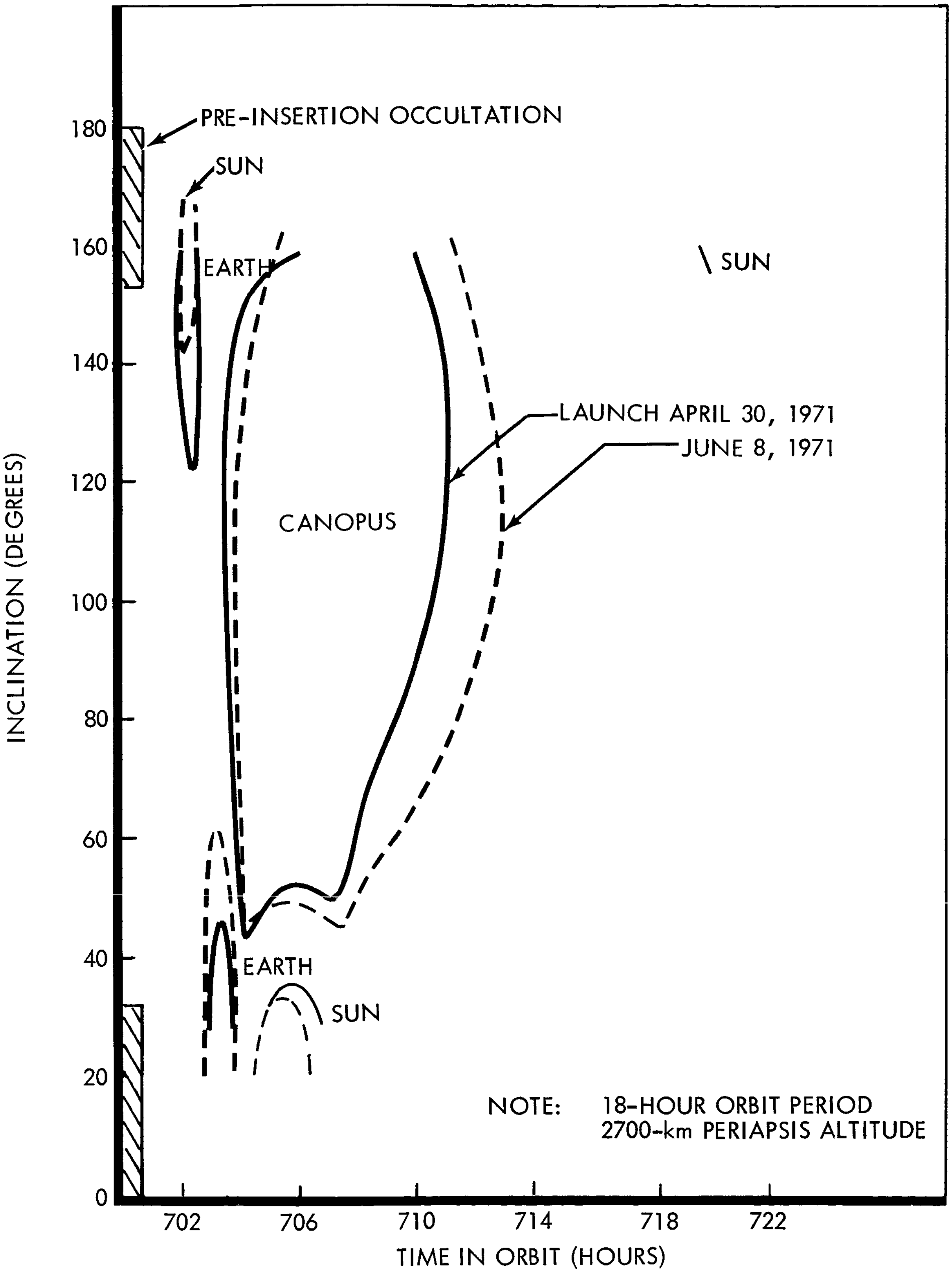


Figure 3.1-25: Occultation in Orbit — 40th Orbit (30th Day)

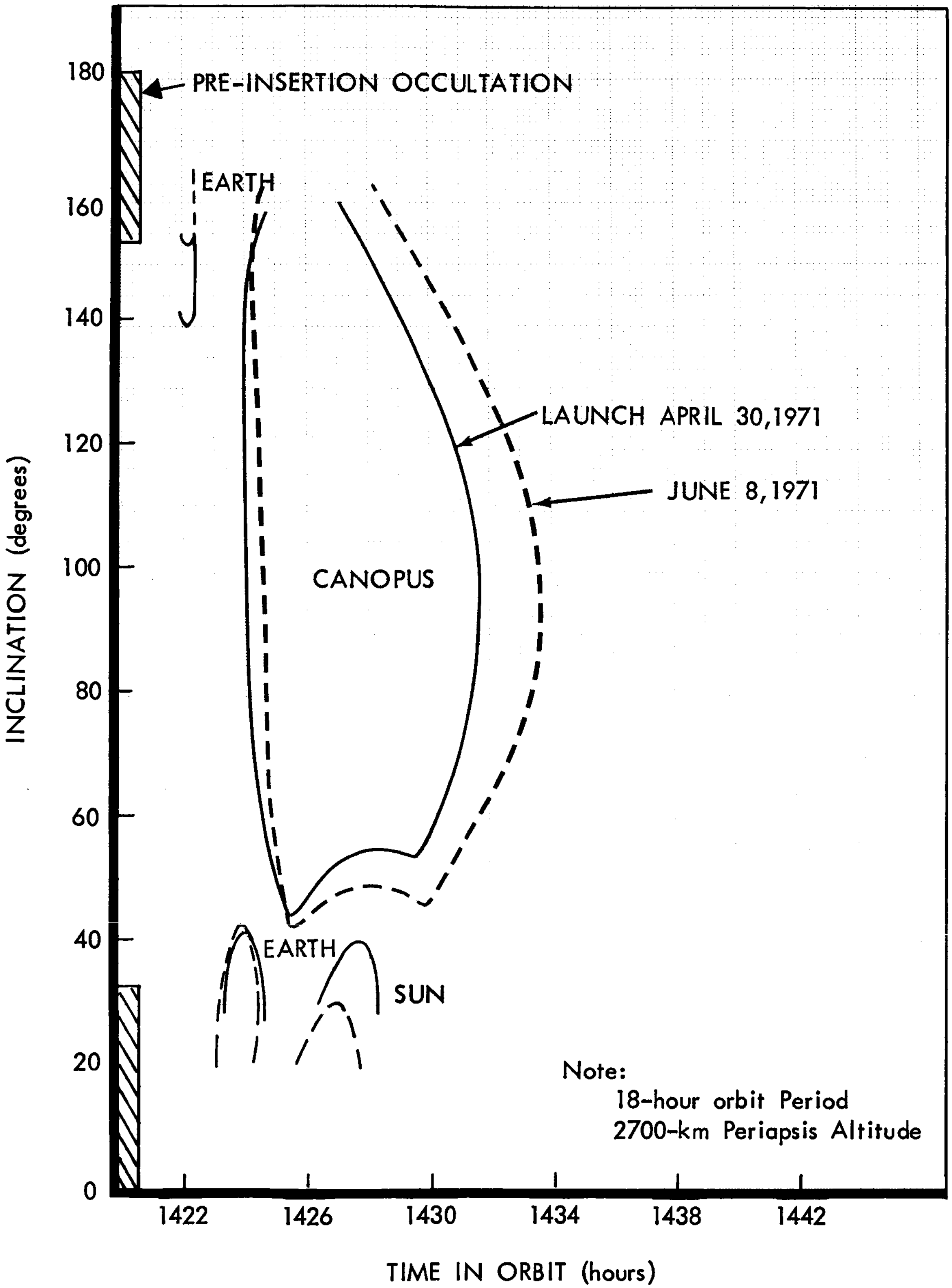


Figure 3.1-26: Occultation in Orbit — 80th Orbit (60th Day)

the near limb of Mars comes within 60 degrees of the spacecraft-Canopus line of sight. Orbits with inclinations between 39 and 42 degrees are free from Canopus and solar occultation for the first 80 orbits.

3.1.4.4 Periapsis Location

The natural periapsis location of the approach hyperbola is fixed by the selection of a particular transit trajectory. The orientation of periapsis may be changed by insertion early or late on the ellipse. Figure 3.1-27 shows an example of this variation of the angle between the approach asymptote and periapsis for tangential insertion into the orbit about Mars at various approach speeds, V_∞. The nominal insertion ΔV for this maneuver is 5700 fps. Early orbit insertion will be used to maintain a constant insertion ΔV while providing a 10-day separation between the two spacecraft. The 10-day separation can be maintained if the second spacecraft uses a trajectory with a constant V_∞ = 3.25 kilometers per second. The angle between the approach asymptote and periapsis will change from 68 to 38 degrees. The approach asymptote will rotate 10 degrees in inertial space and the Sun will move 6 degrees. The resulting illumination angle is decreased by 14 degrees at periapsis for the second spacecraft.

3.1.4.5 Detailed Mars Orbit Characteristics

Orbit Insertion--Figure 3.1-28 shows the impulsive velocity requirements for insertion into the Mars orbit. Various periapsis altitudes and orbit periods are shown for an approach velocity of 3.5 kilometers per second. Figure 3.1-29 shows the ΔV requirements adjusted to compensate for

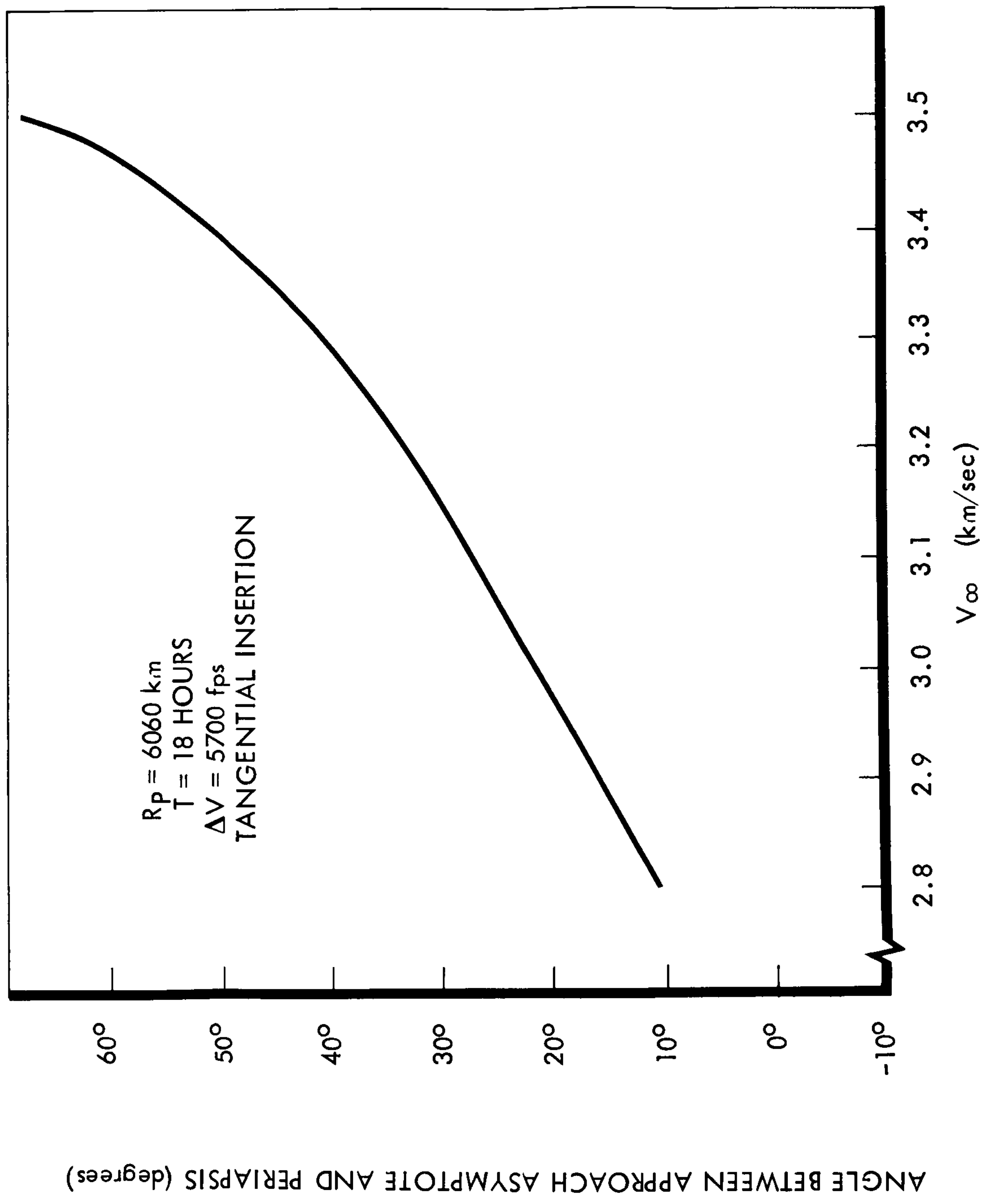


Figure 3.1-27: Variation of Periapsis Position

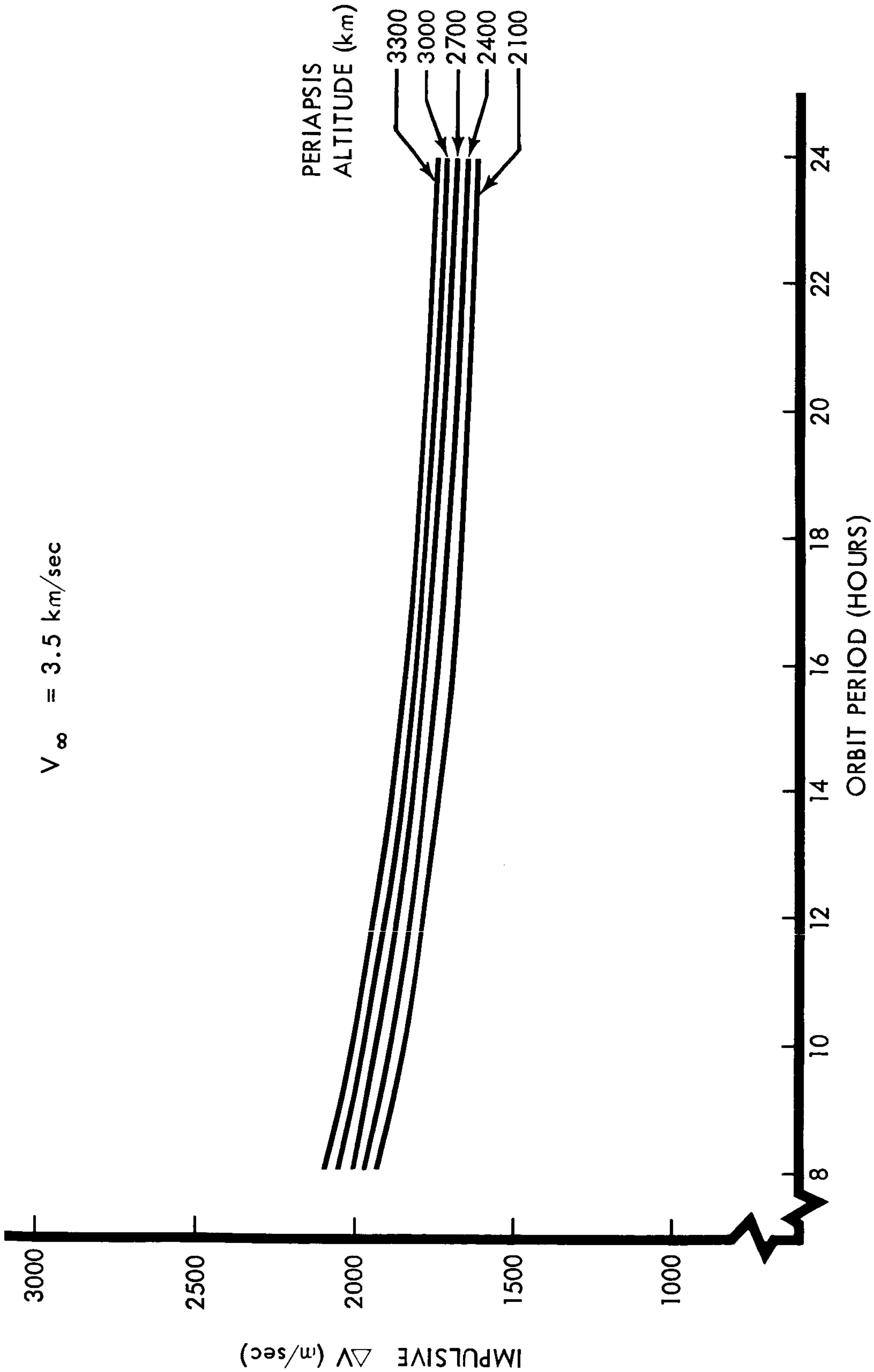


Figure 3.1-28: Insertion Velocity Requirements

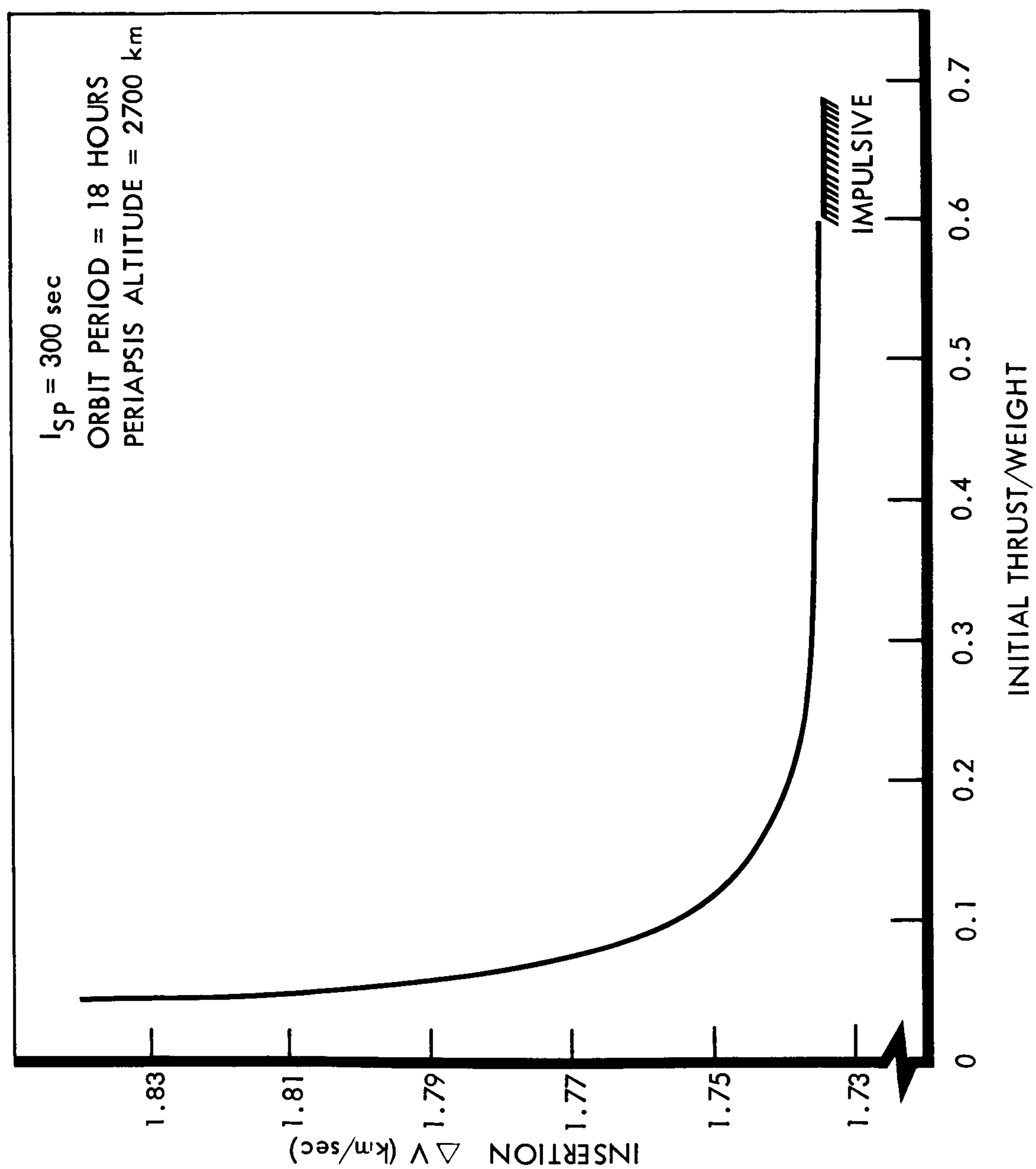


Figure 3.1-29: Effect of Finite Thrust at Insertion

finite thrusting as a function of the initial thrust-to-weight ratio. The data are for a gravity-turn trajectory (velocity vector and thrust vector always aligned) thrusting into an 18-hour period orbit. Gravity turn trajectories must rely on a complicated guidance control law. Thrusting in a fixed inertial direction has therefore been considered as well. Figure 3.1-30 shows the ΔV losses for this thrusting mode as a function of the inertial thrust angle for two different thrust-to-weight ratios. The inertial thrust angle is defined in the diagram on the figure. For an initial thrust-to-weight ratio of 0.11, thrusting in an optimized inertially fixed direction will increase the required ΔV by only 3 meters per second over that for a gravity-turn trajectory.

The dispersions in the periapsis altitude resulting from errors in the thrust angle are shown in Figure 3.1-31. The periapsis altitude is relatively insensitive to small errors in the thrust direction. Figure 3.1-32 shows the dispersions in orbital period that result from an error in the insertion velocity impulse. For this example, the orbit period is extremely sensitive to errors in the velocity impulse.

<u>Orbiting Trajectory</u>--To show the orbit trajectory characteristics in detail, a specific orbit has been selected. This orbit has an inclination of approximately 40° to the Mars equator, a period of 18 hours, and a periapsis altitude of 2700 kilometers. Figure 3.1-33 shows the right ascension and declination of the spacecraft during the first and eightieth orbits. The slight change in position of the orbit is caused by the regression of the nodal longitude and precession of the argument

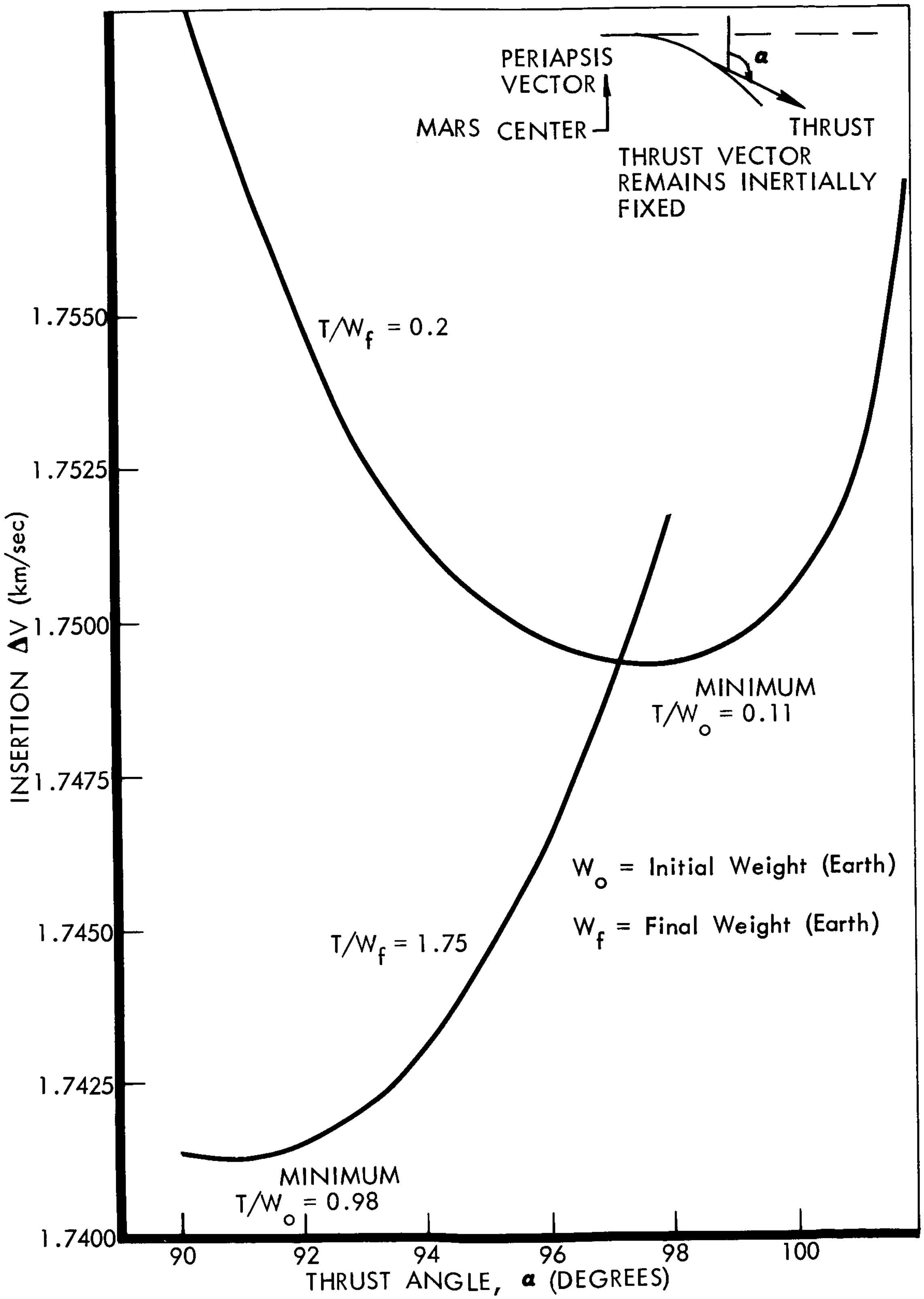


Figure 3.1-30: Velocity Requirements for Thrusting in an Inertially Fixed Direction

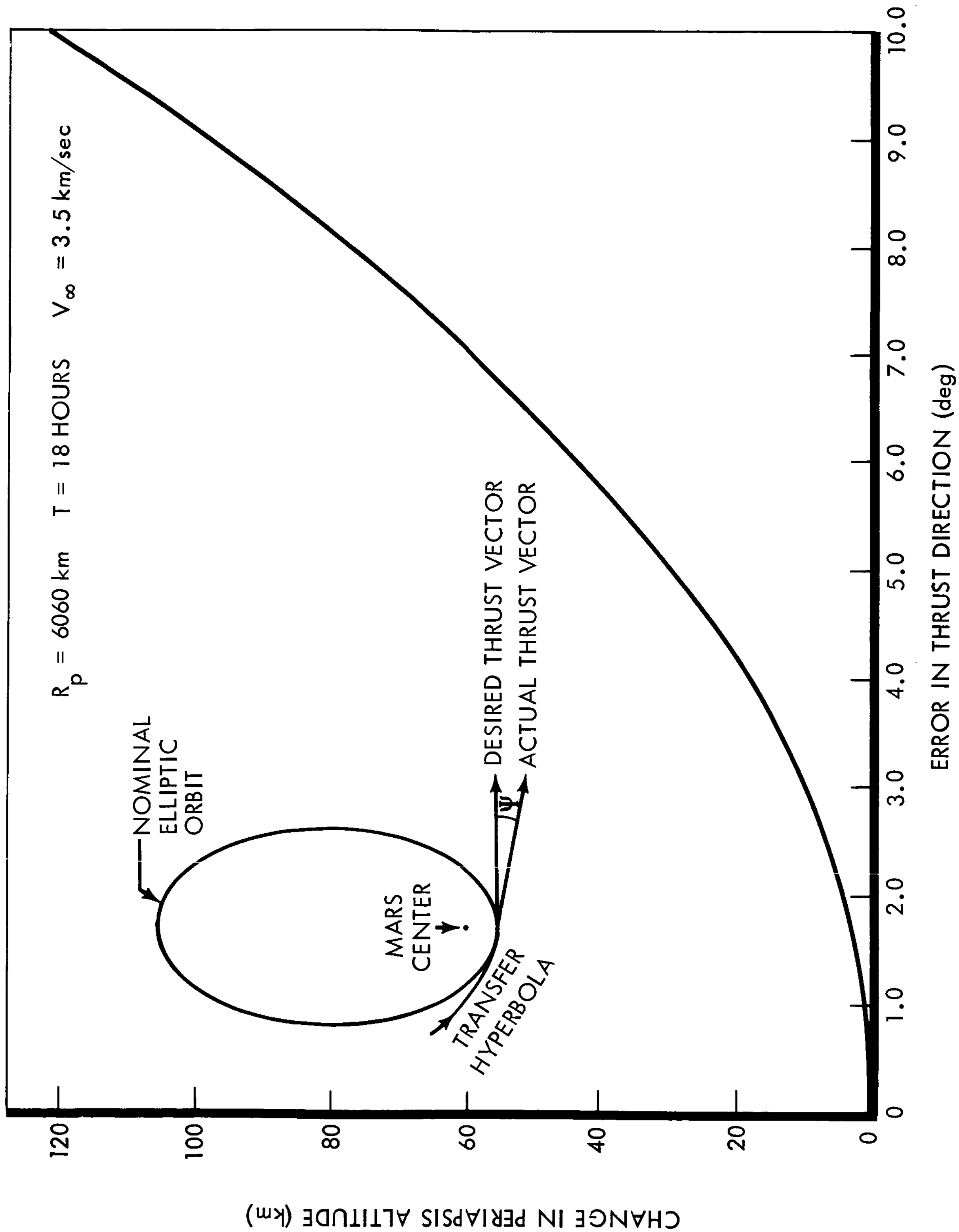


Figure 3.1-31: Periapsis Sensitivity for Orbital Insertion at Periapsis

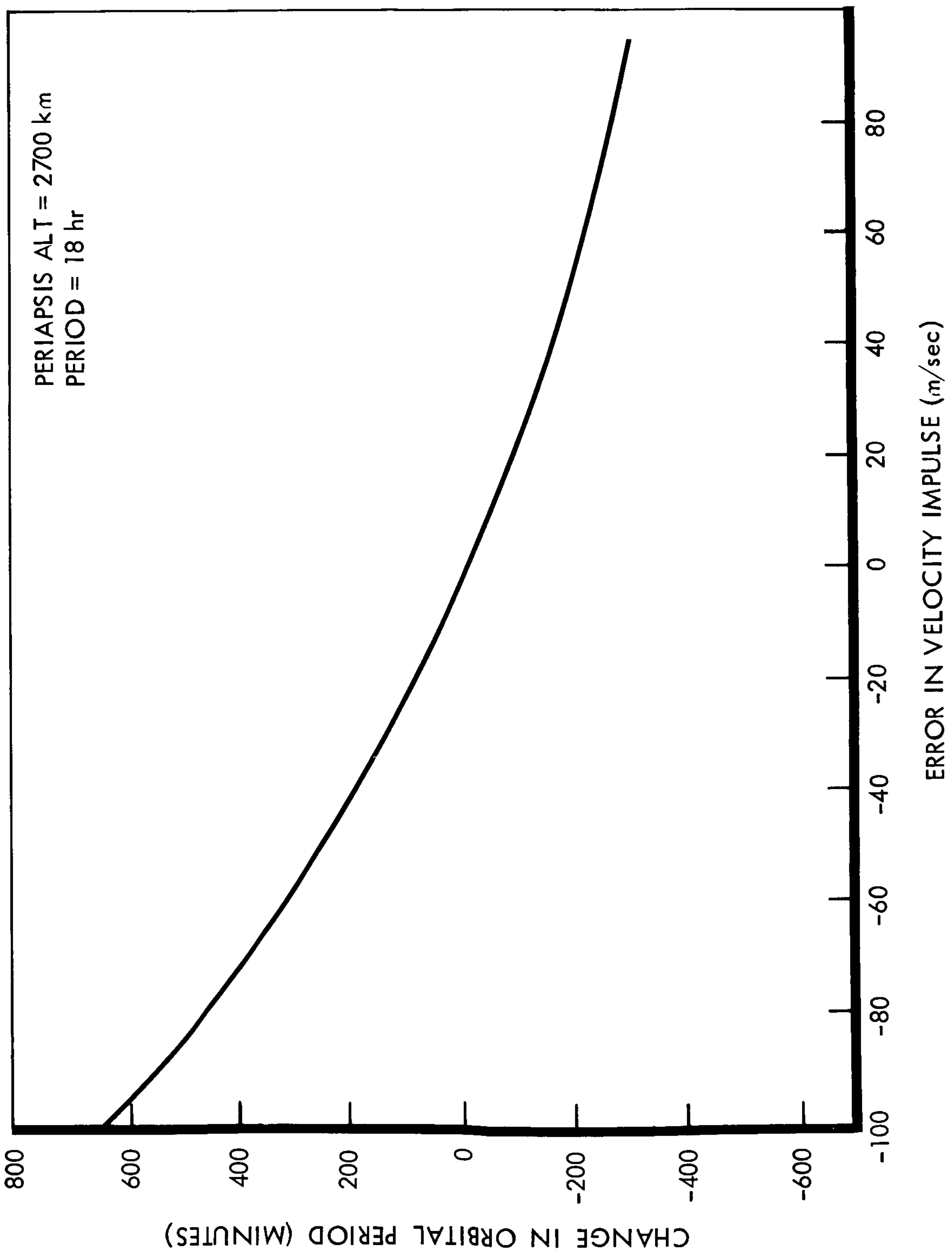


Figure 3.1-32: Period Sensitivity for Orbital Insertion at Periapsis

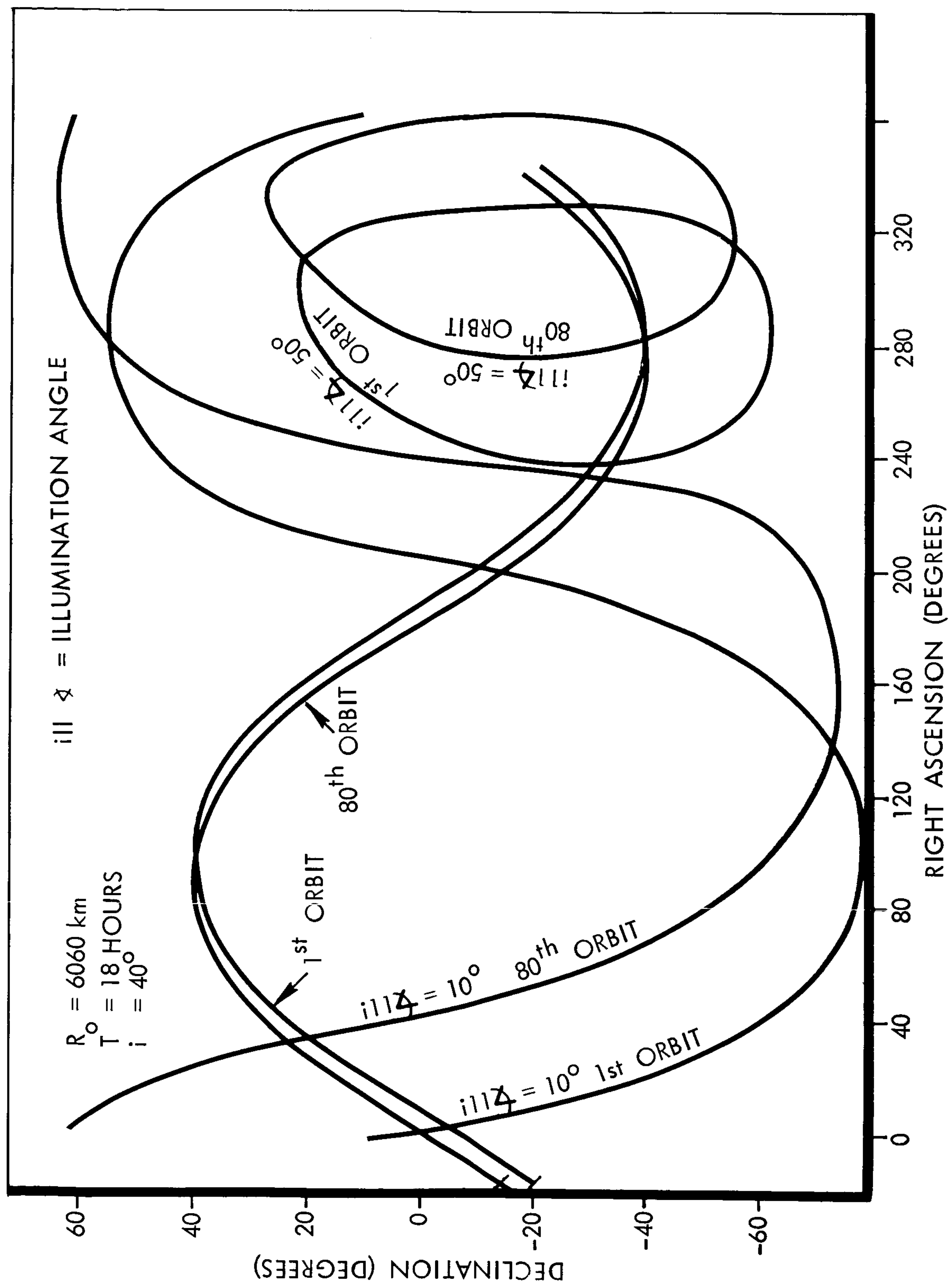


Figure 3.1-33: Declination vs Right Ascension of Satellite ; Illumination Angle Crossplots

of periapsis due to the oblateness of Mars. Also shown on the figure are lines of constant illumination angle (measured from the terminator) for 10 and 50 degrees. The change in position of these lines is due to the motion of Mars about the Sun. As the mission progresses, the illumination angle at periapsis gets larger. Therefore the periapsis point begins as a good place for black-and-white television, and progresses to a situation favoring color television. Figures 3.1-34 through 3.1-36 show the latitudes and illumination angles for the first, fortieth, and eightieth orbits. The two curves shown on each figure represent the maximum variation in the location of the approach asymptote for the V_∞ = 3.5 kilometers per second trajectory set. These limits occur for launch on April 30 and June 8. The favorable region of latitudes from +10 to -40 degrees and illumination angles from 10 to 50 degrees are indicated by the shaded rectangular area on each chart. For this example, all of the first 80 orbits have favorable illumination angles and latitude coverage near periapsis.

Figure 3.1-37 shows a ground track on a mercator projection of Mars for the first, fortieth, and eightieth orbits. The changes in shape are due to the cumulative effects of orbit perturbations from Mars' oblate gravity field.

Longitude adjustments of these paths can be made at Earth departure by varying the time of initial encounter at Mars by ±12-hours. Some limitations exist on this adjustment, however, if encounter is to take place in view of the Goldstone DSN station.

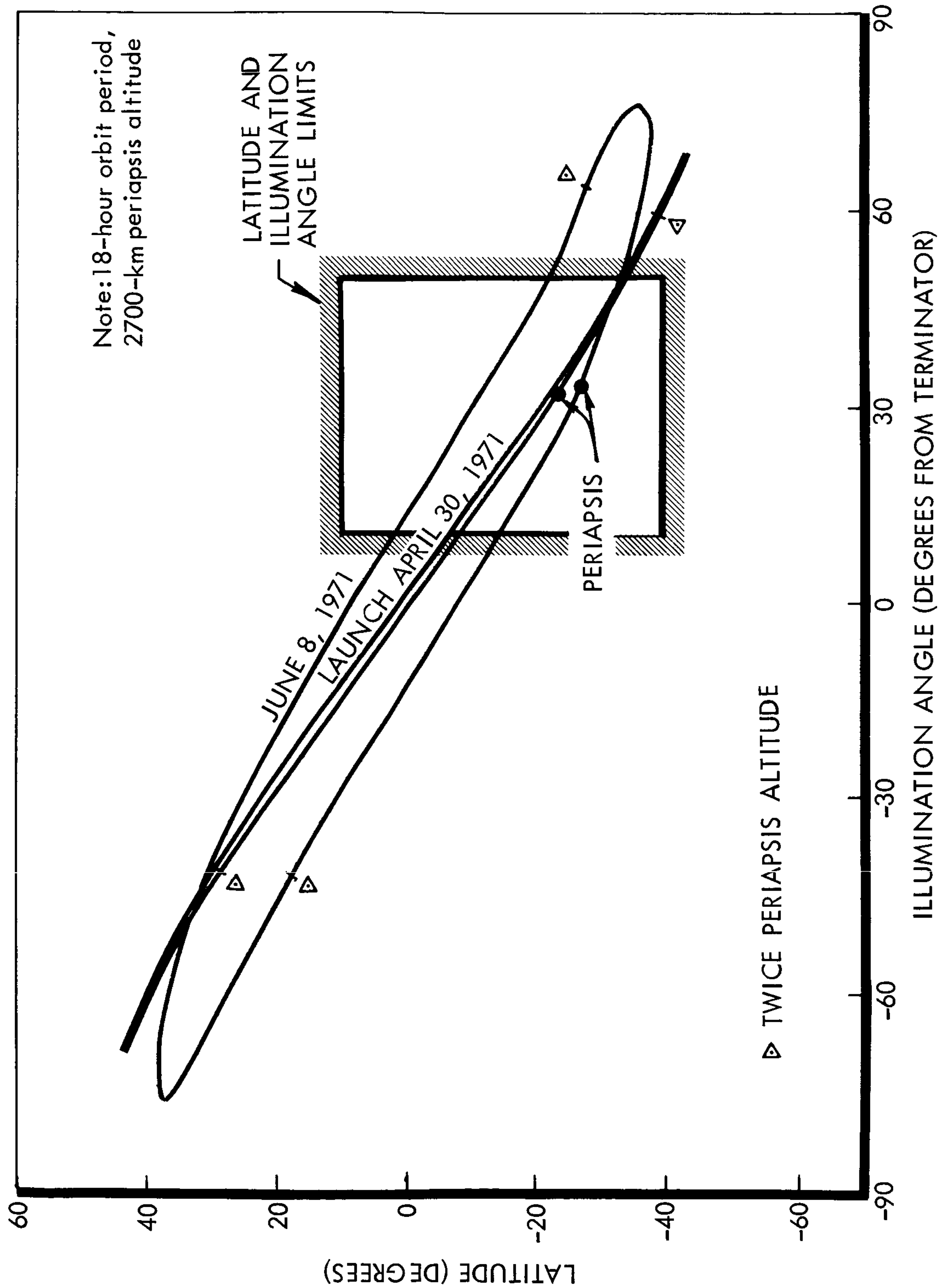


Figure 3.1-34: Altitude-Position-Lighting Relation — First Orbit (1st Day)

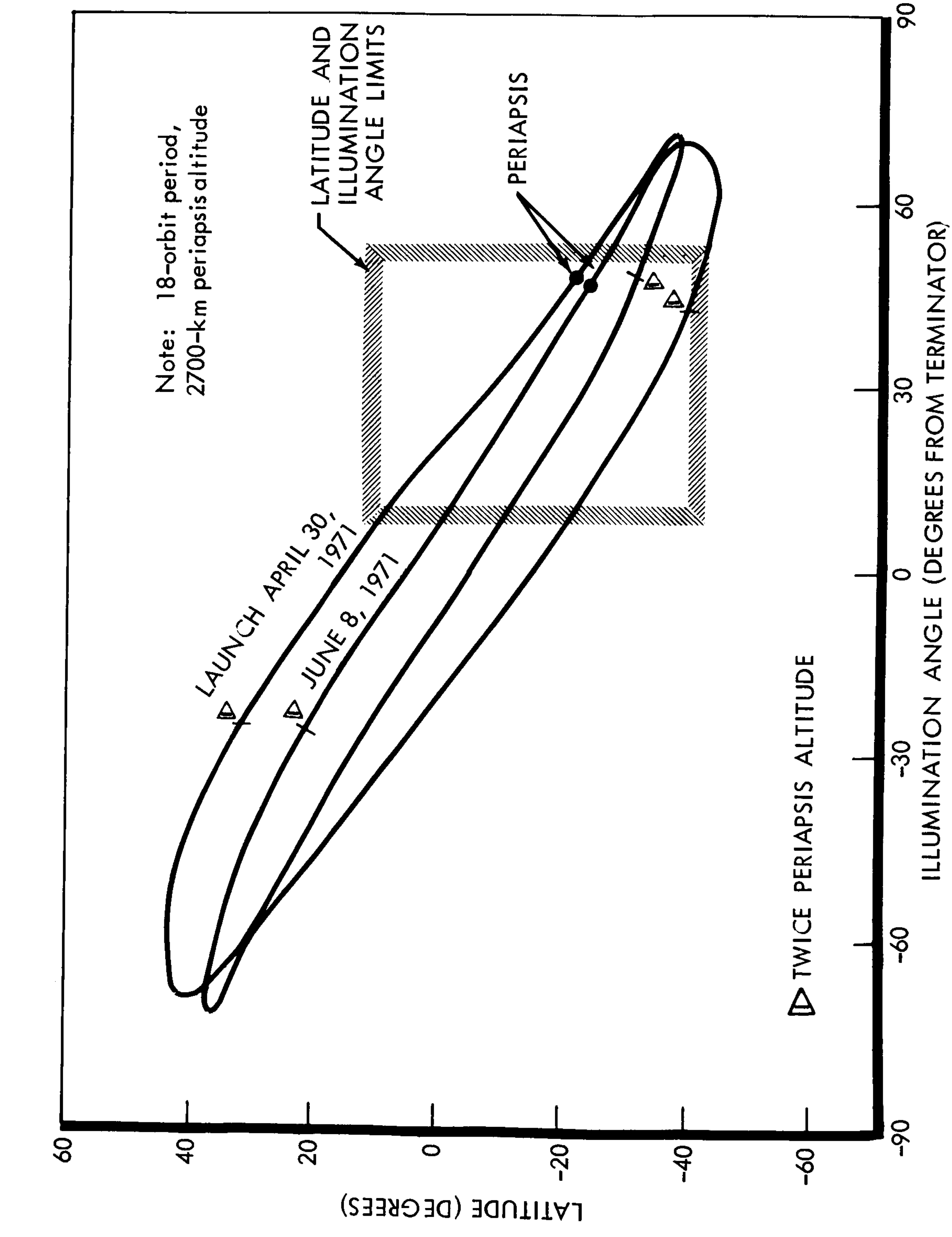


Figure 3.1-35: Altitude-Position-Lighting Relation — 40th Orbit (30th Day)

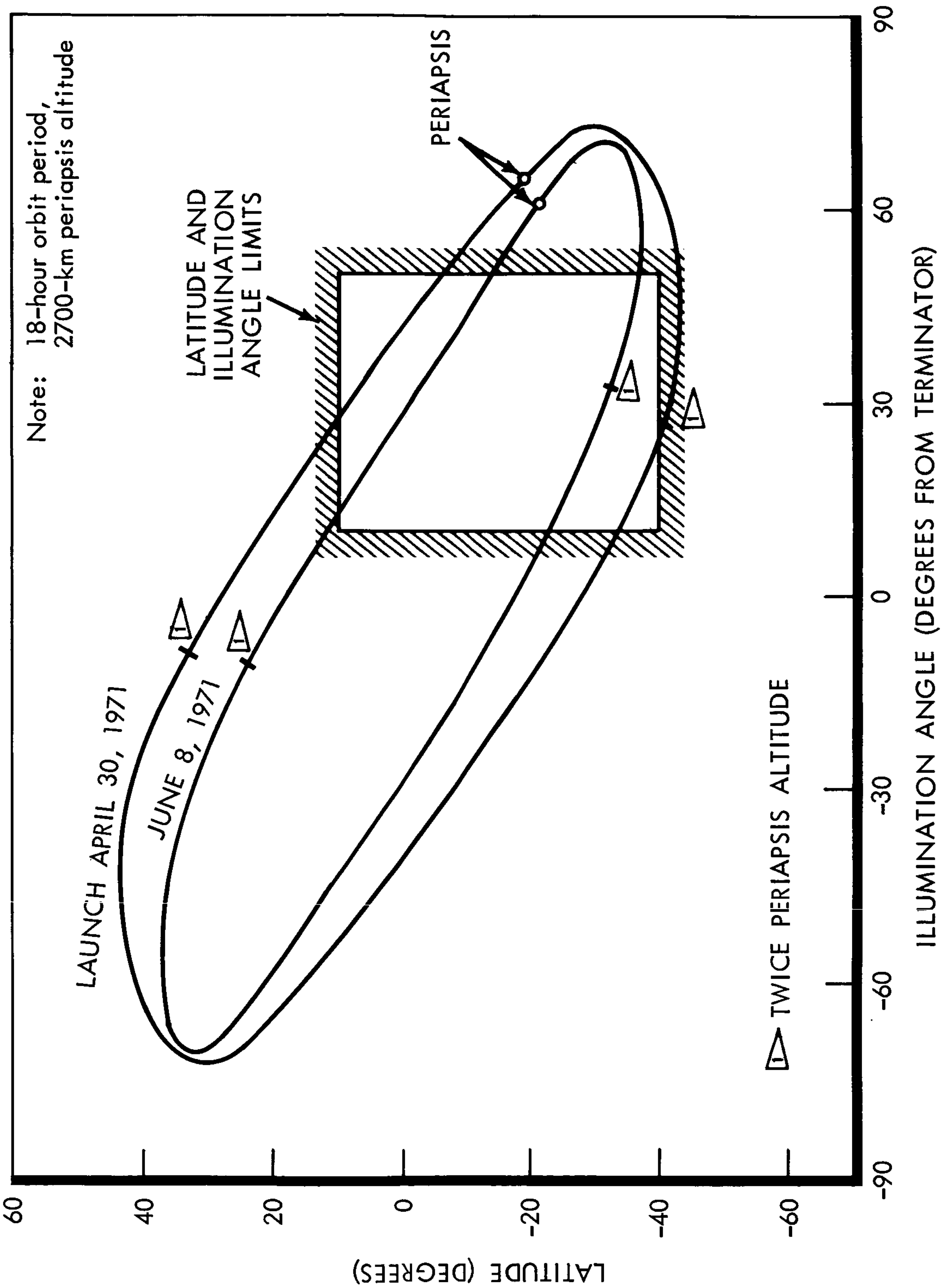
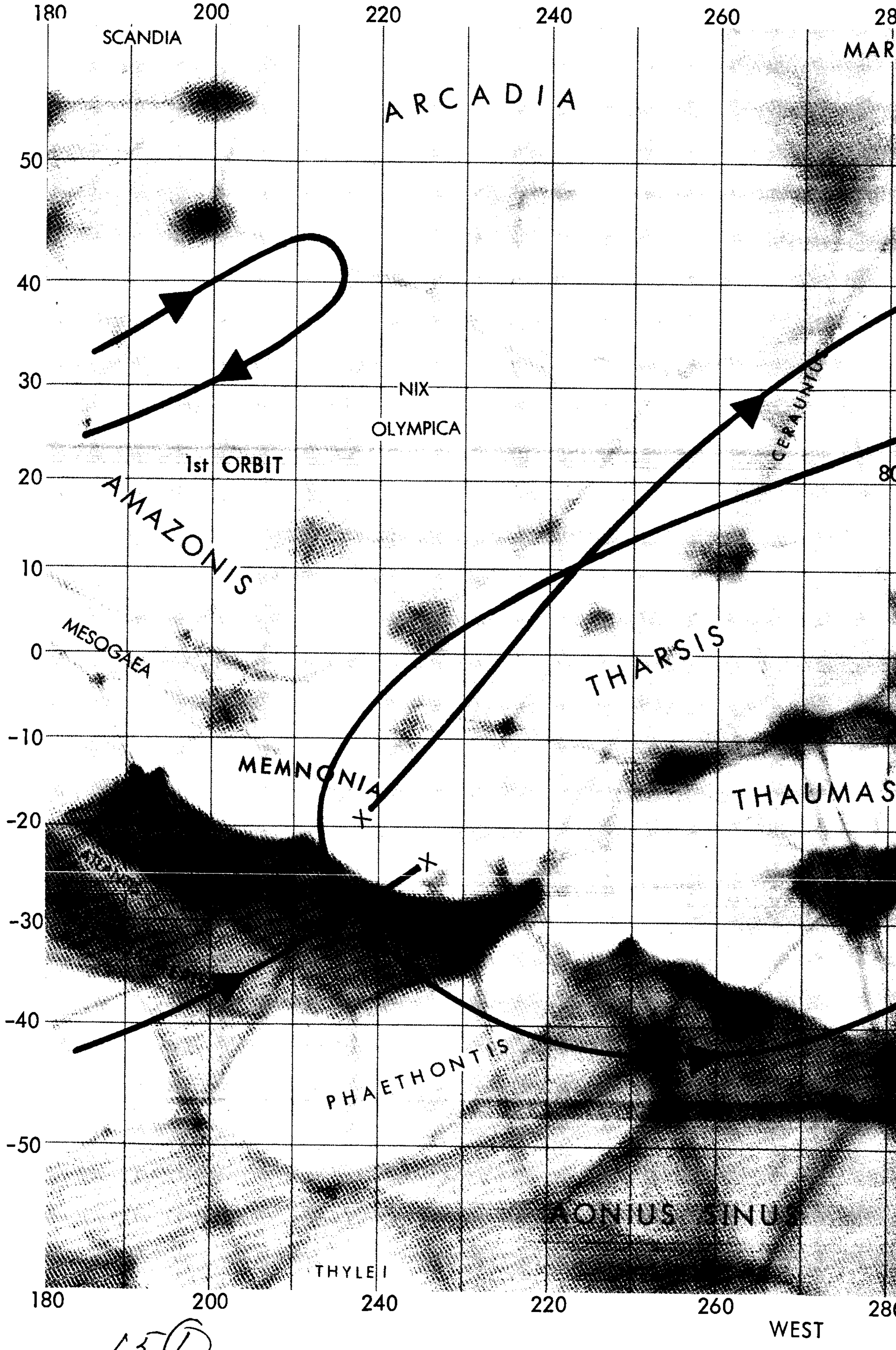


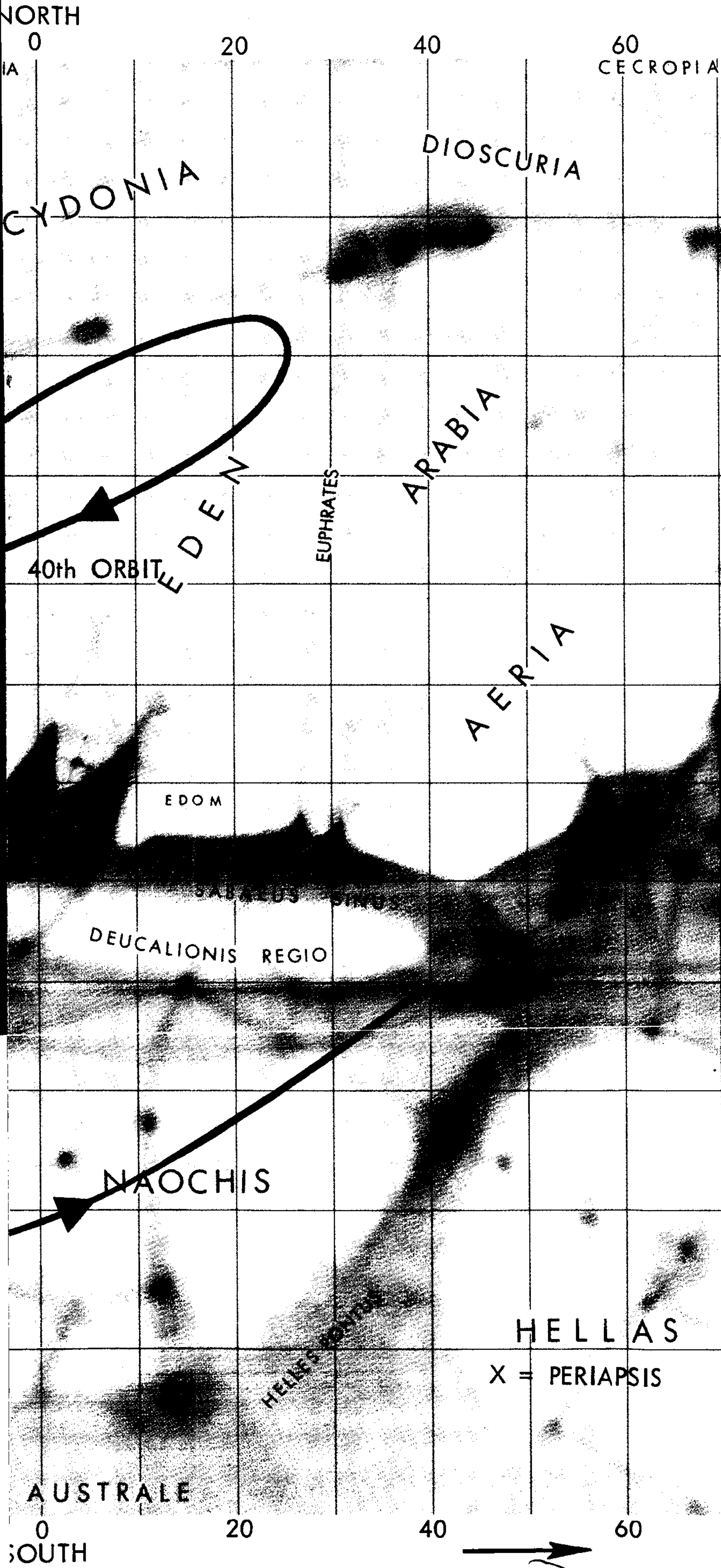
Figure 3.1-36: Altitude-Position-Lighting Relation — 80th Orbit (60th Day)


180
200
220
240
260
SCANDIA
ARCADIA
50
40
30
NIX
OLYMPICA
1st ORBIT
20
AMAZONIS
10
MESOGAEA
0
THARSIS
-10
MEMNONIA
-20
-30
-40
PHAETHONTIS
-50
AONIUS SINUS
THYLE I
180
200
240
220
260
WEST

65 (1)

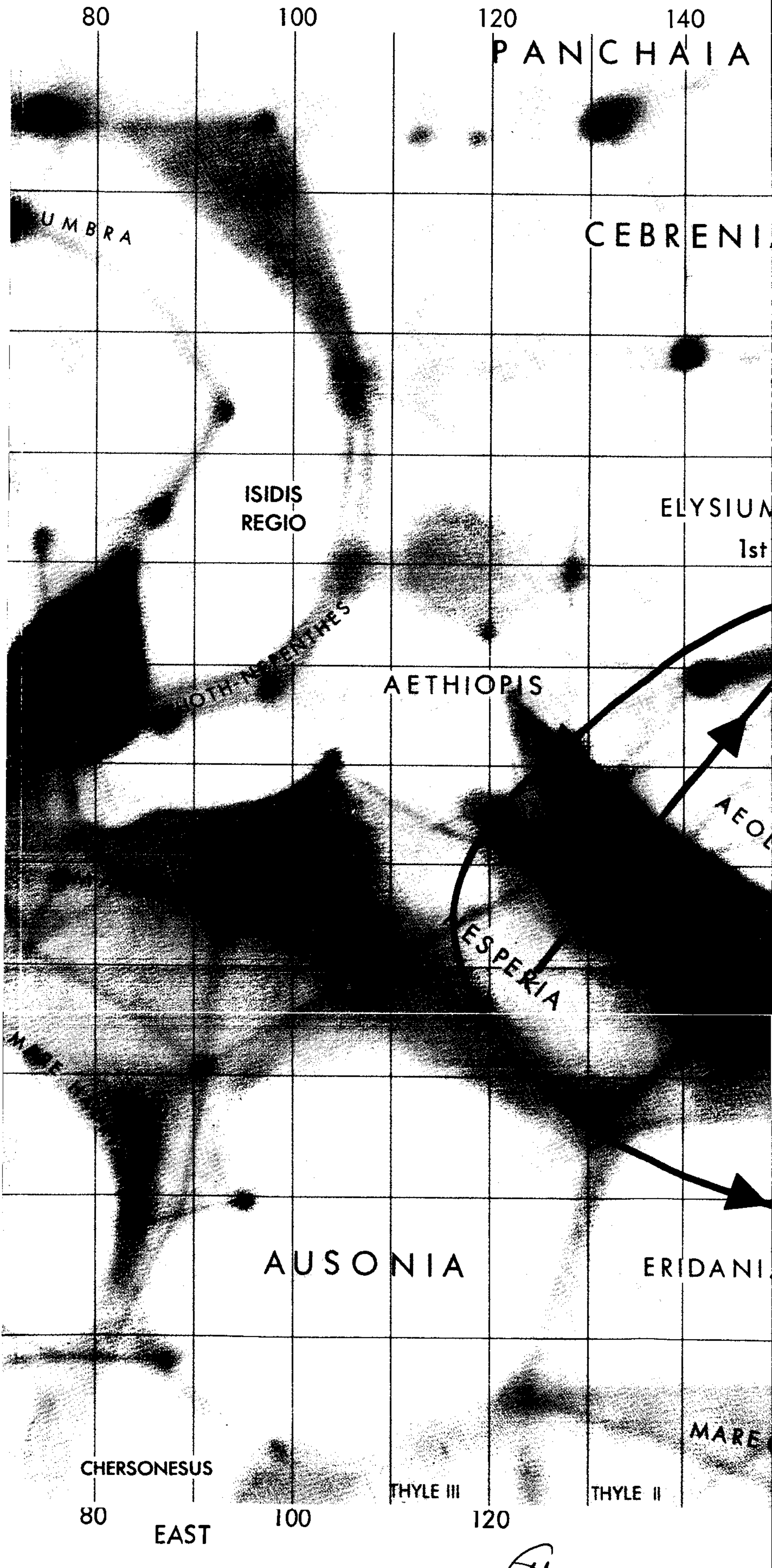
300
320
340
BOREUM
ORTYG
TEMPE
th ORBIT
XANTHE
CHRYSE
CANDOR
IA
ARGYRE I
OGYGIS
REGIO
MARE
300
320
340

65 2



NORTH
0
20
40
60
CECROPIA
DIOSCURIA
CYDONIA
ARABIA
EDEN
EUPHRATES
40th ORBIT
AERIA
EDOM
DEUCALIONIS REGIO
NAOCHIS
HELLAS
X = PERIAPSIS
AUSTRALE
0
20
40
60
SOUTH

65 (3)



80
100
120
140
PANCHAIA
UMBRA
CEBRENI
ISIDIS
REGIO
ELYSIU
1st
AETHIOPIS
ESPERIA
AUSONIA
ERIDANI
CHERSONESUS
THYLE III
THYLE II
80
EAST
100
120

65

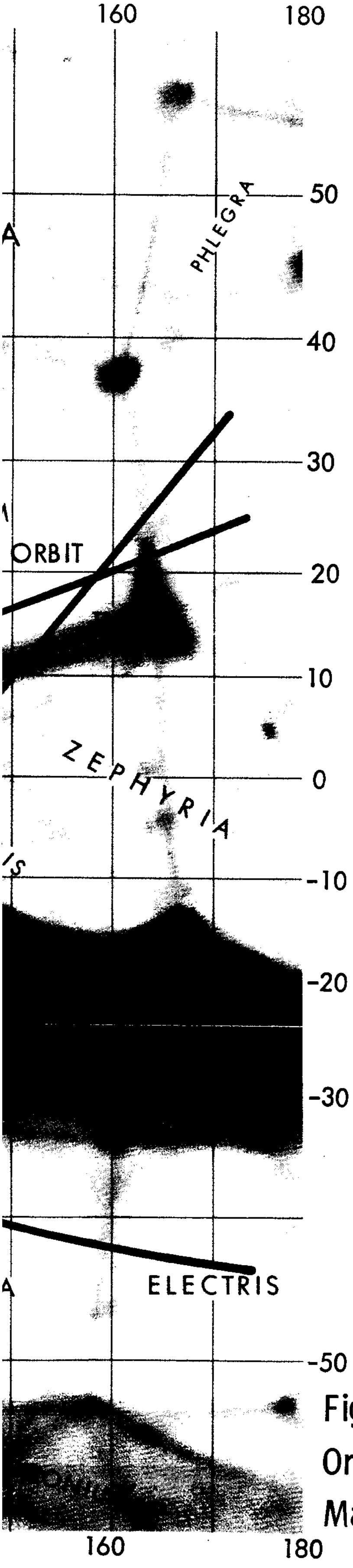


Figure 3.1-37: Orbit Path on Mars Surface

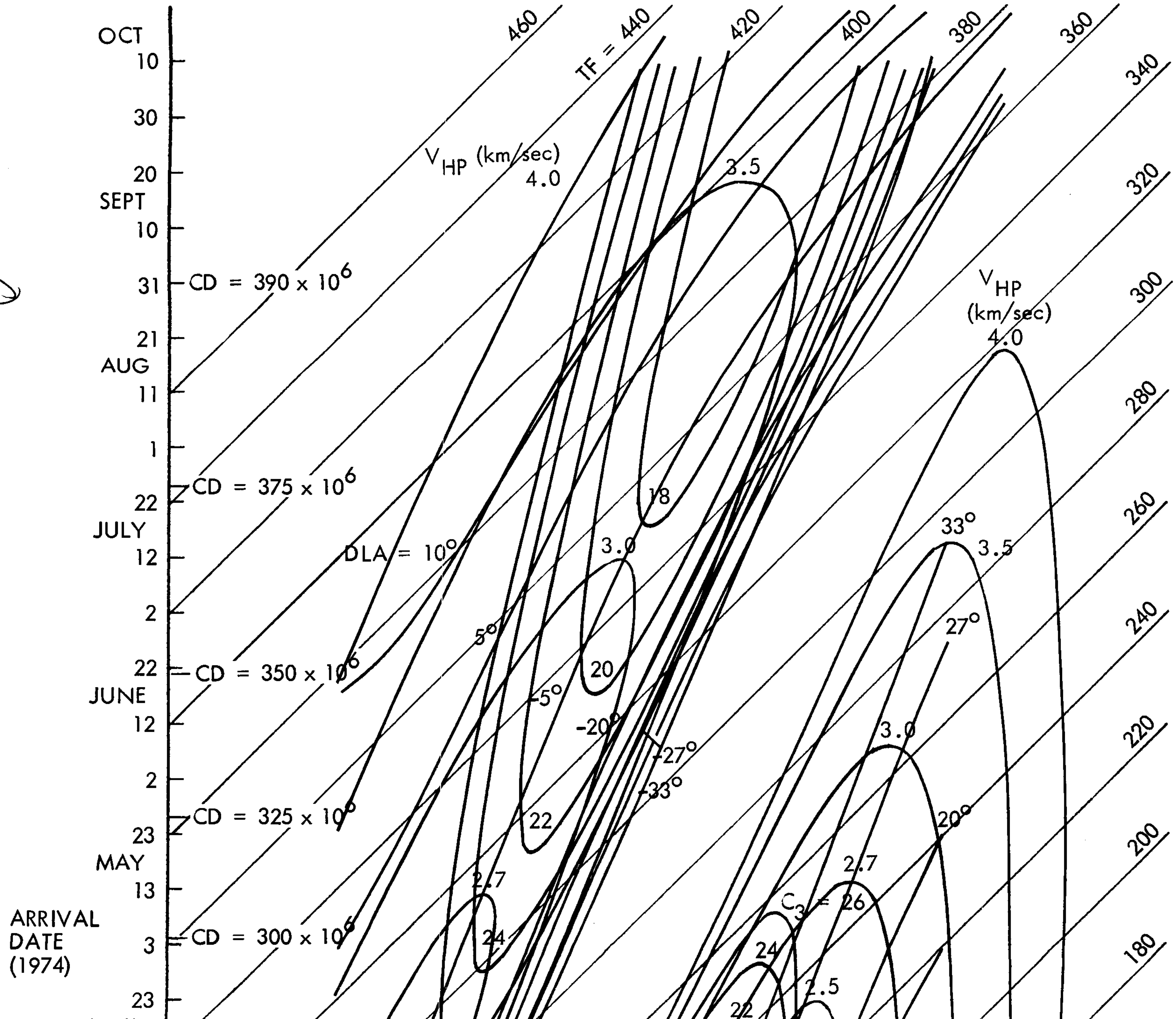
Orbit Corrections--After the spacecraft has been in orbit for several days, sufficient information will be available to attempt an adjustment of the orbit parameters. The most important single parameter to be controlled is the orbit period. Figure 3.1-31 can be used to determine the change in orbit period which is available through application of ΔV at periapsis. The spacecraft has 100-meters-per-second orbit trim capability and consequently can increase the orbit period by 10.9 hours or decrease the period by 5.2 hours, from the example 18-hour orbit.

3.1.5 Capability for 1973 and Subsequent Missions

The compatibility with 1973 orbiter missions is discussed in Section 3.1.2. For technical detail, design trajectory charts for 1973 are shown in Figure 3.1-38.

Missions in the 1975 and 1977 opportunities have been examined (in addition to the 1973 mission) to compare the compatibility with the 1971 mission. Figure 3.1-39 shows the basic design chart for the 1975 Type I missions. Curves of constant declination of the geocentric asymptote (DLA) are shown in red. The constant asymptotic approach speed (V_∞) curves are shown in blue. For these missions, the minimum C_3 is 18.7 km^2/sec^2. This is beyond the assumed Saturn IB/Centaur capability and a versatile set of trajectories would require drastically reduced payloads. Type II trajectories are therefore more preferable for large payloads. Figure 3.1-40 shows these Type II trajectories with the DLA and V_∞ curves superimposed. The DSN tracking limits of $\pm$5 degrees are shown by the red shaded region. The grey shaded region represents the limiting booster capability of 18 km^2/sec^2. A wide range of missions are available during the opportunity. A launch is possible any day from August 20, 1975, to October 25, 1975, with arrival from August 4, 1976, through December 22, 1976. If a decision is made to use an orbiting spacecraft in the 1975 mission, the V_∞ = 3.5-kilometers-per-second line shows sufficient ΔV available from the propulsion system to perform orbiter missions similar to the 1971 mission by launching between August 20, 1975, and October 14, 1975. Figure 3.1-41, the basic design chart for Type I missions in the 1977 era, also indicates excessively high C_3 energies are required to meet the present DLA constraints with a versatile mission capability. In Figure 3.1-42 (Type II trajectories for 1977) there is a wide range of trajectories available for flyby missions launched between September 6, 1977, and at least December 2, 1977. Orbiter missions similar to 1971 could, if desired, be flown in 1977, with launches between September 6, 1977, and November 8, 1977. For completeness, the 1971 trajectory design chart is presented in Figure 3.1-43.

69 1



OCT
10
30
20
SEPT
10
31
21
AUG
11
1
22
JULY
12
2
22
JUNE
12
2
23
MAY
13
3
23
ARRIVAL DATE (1974)
CD = 390 x 10^6
CD = 375 x 10^6
CD = 350 x 10^6
CD = 325 x 10^6
CD = 300 x 10^6
TF = 440
460
420
400
380
360
340
320
300
280
260
240
220
200
180
V_HP (km/sec)
4.0
3.5
3.0
2.7
2.5
DLA = 10°
5°
-5°
-20°
-27°
-33°
33°
27°
20°
C_3 = 26
24
22
20
18

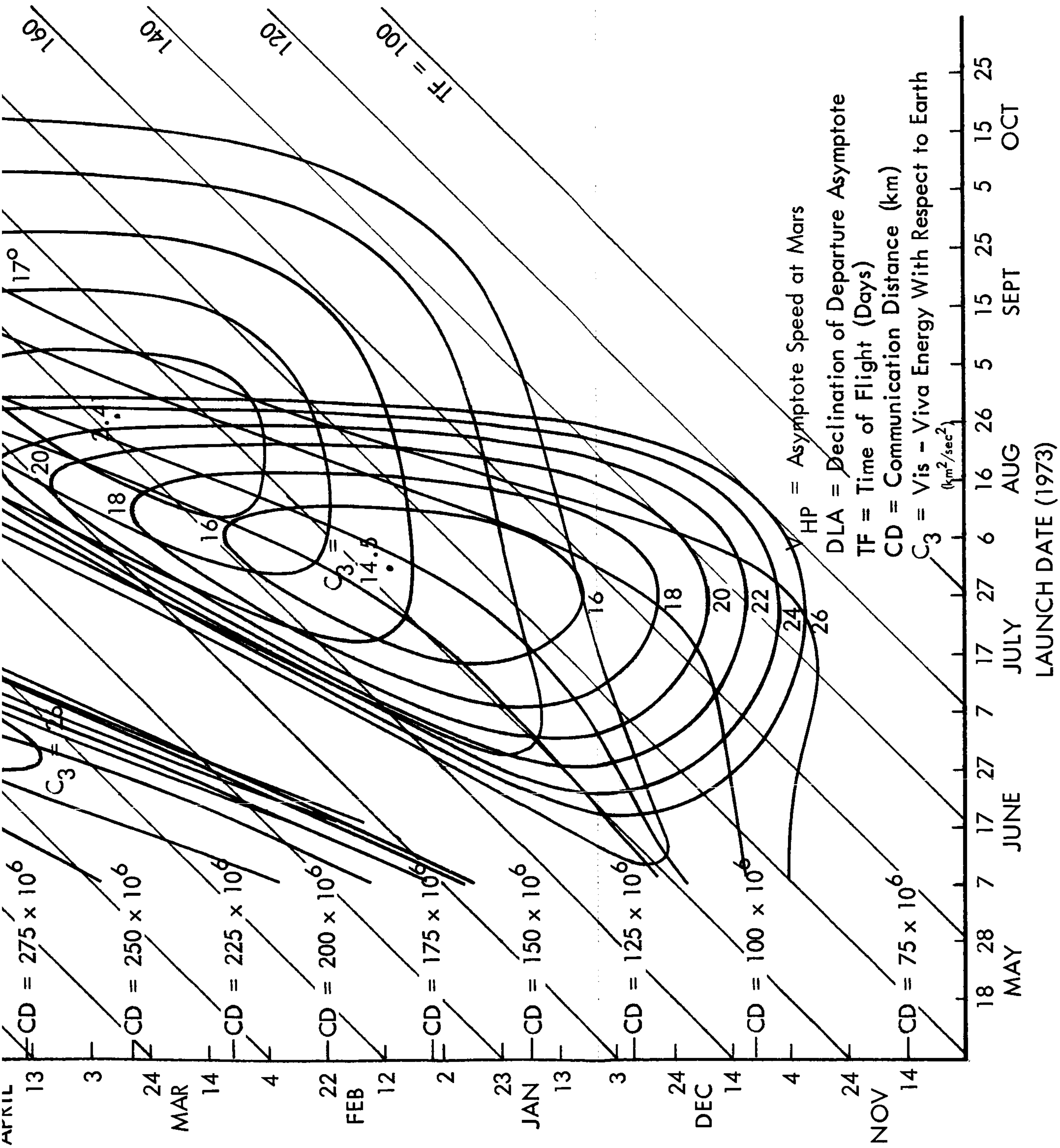
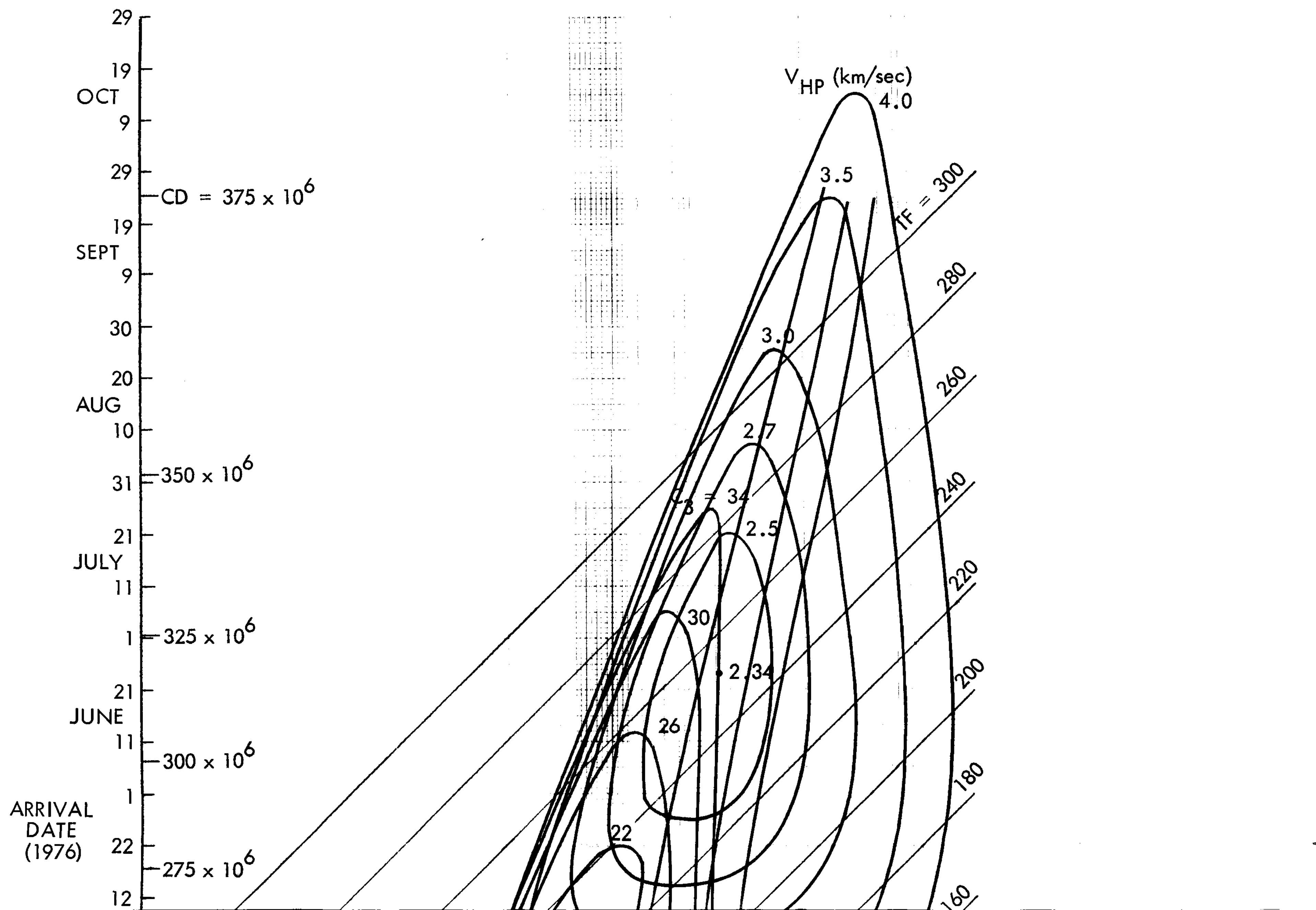


Figure 3.1-38: Earth-Mars Transfer Characteristics — 1973 Types I and II


ARRIVAL
DATE
(1976)
OCT
SEPT
AUG
JULY
JUNE
29
19
9
29
19
9
30
20
10
31
21
11
1
21
11
1
22
12
CD = 375 x 10^6
350 x 10^6
325 x 10^6
300 x 10^6
275 x 10^6
V_HP (km/sec)
4.0
3.5
3.0
2.7
2.5
2.34
C_3 = 34
30
26
22
TF = 300
280
260
240
220
200
180
160

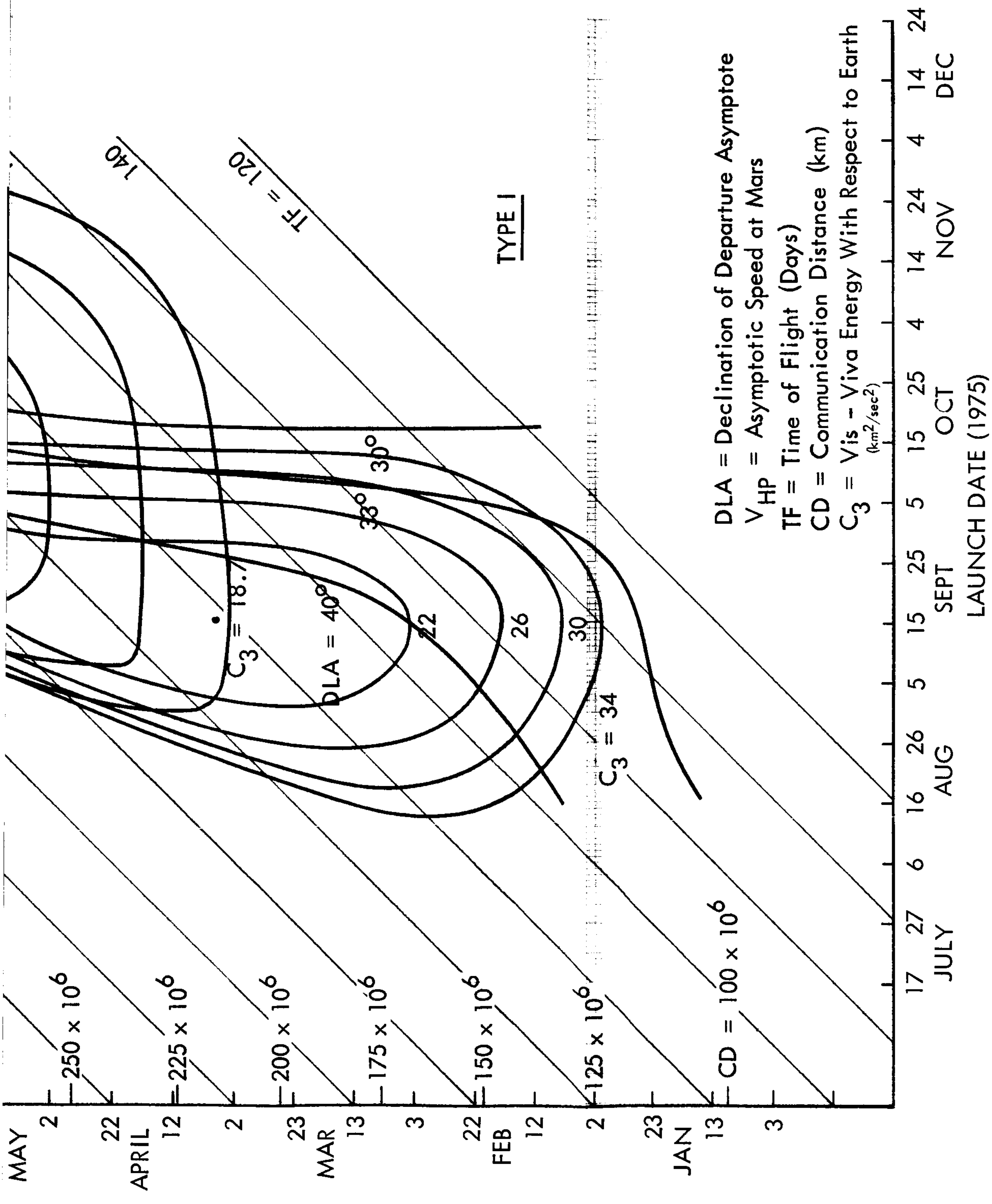
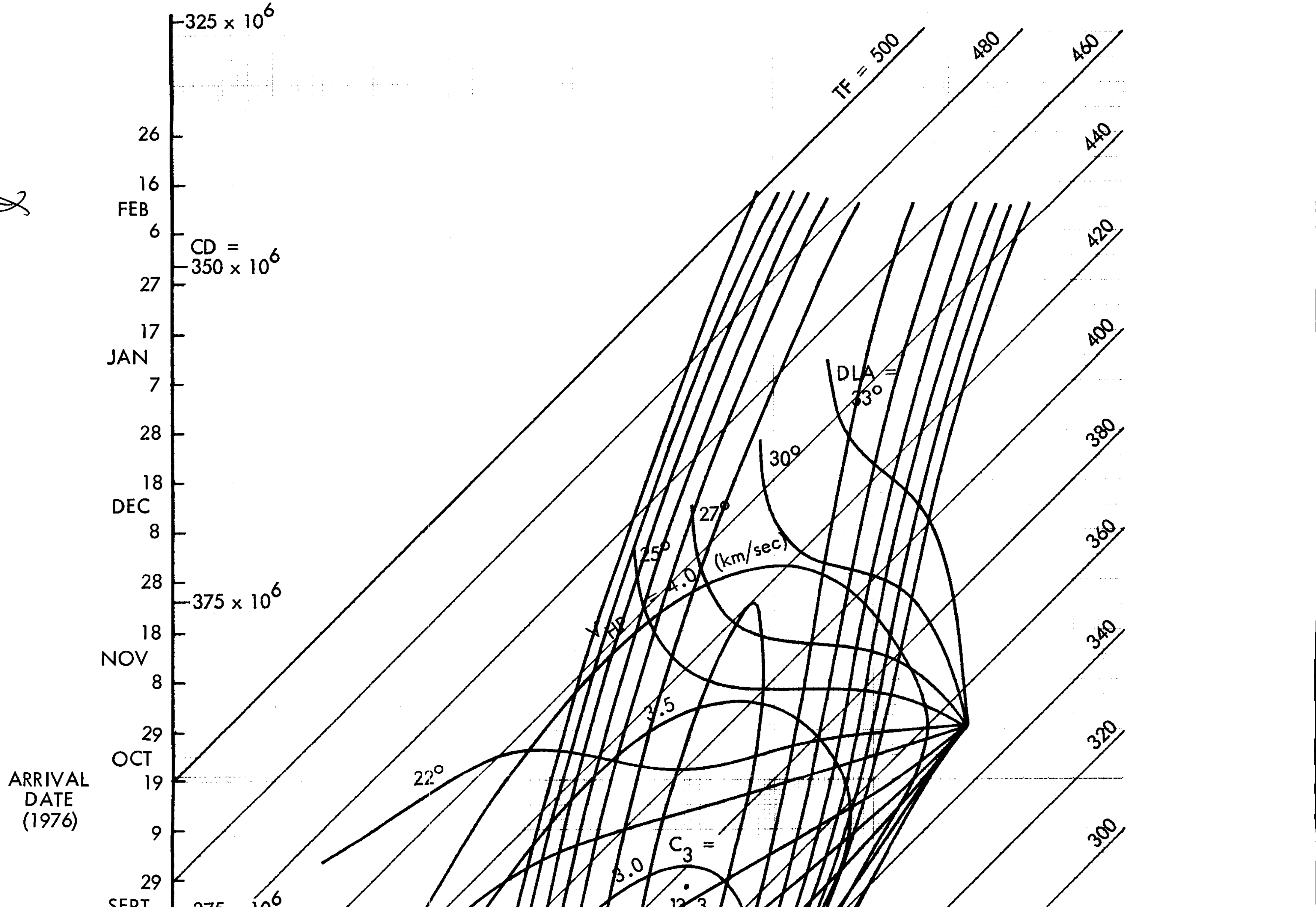


Figure 3.1-39: Earth-Mars Transfer Characteristics — 1975 Type I

2



325 x 10^6
CD =
350 x 10^6
375 x 10^6
26
16
FEB
6
27
17
JAN
7
28
18
DEC
8
28
18
NOV
8
29
OCT
19
9
29
ARRIVAL
DATE
(1976)
TF = 500
480
460
440
420
400
380
360
340
320
300
DLA =
33°
30°
27°
25°
22°
(km/sec)
4.0
3.5
3.0
C3 =

730

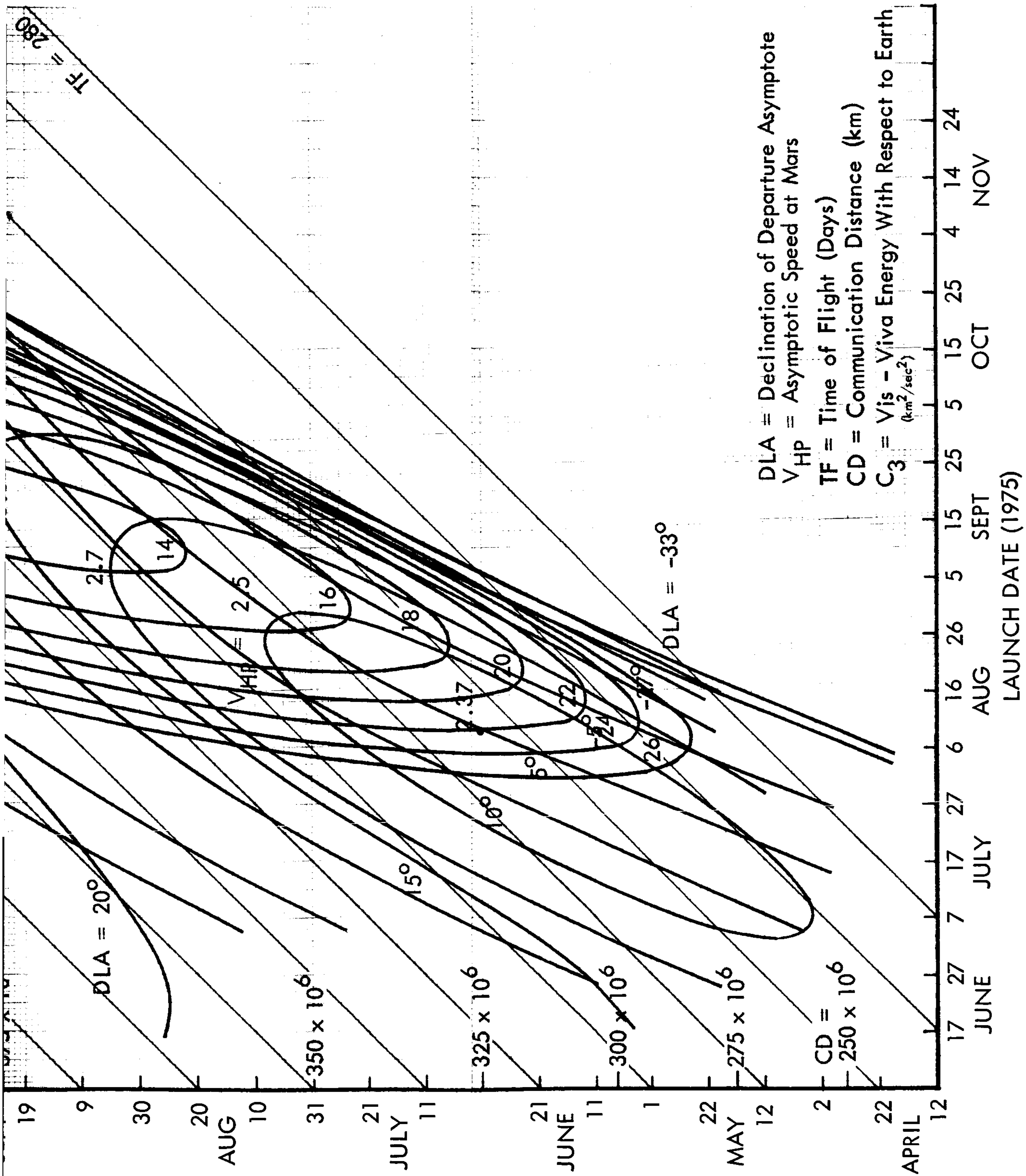
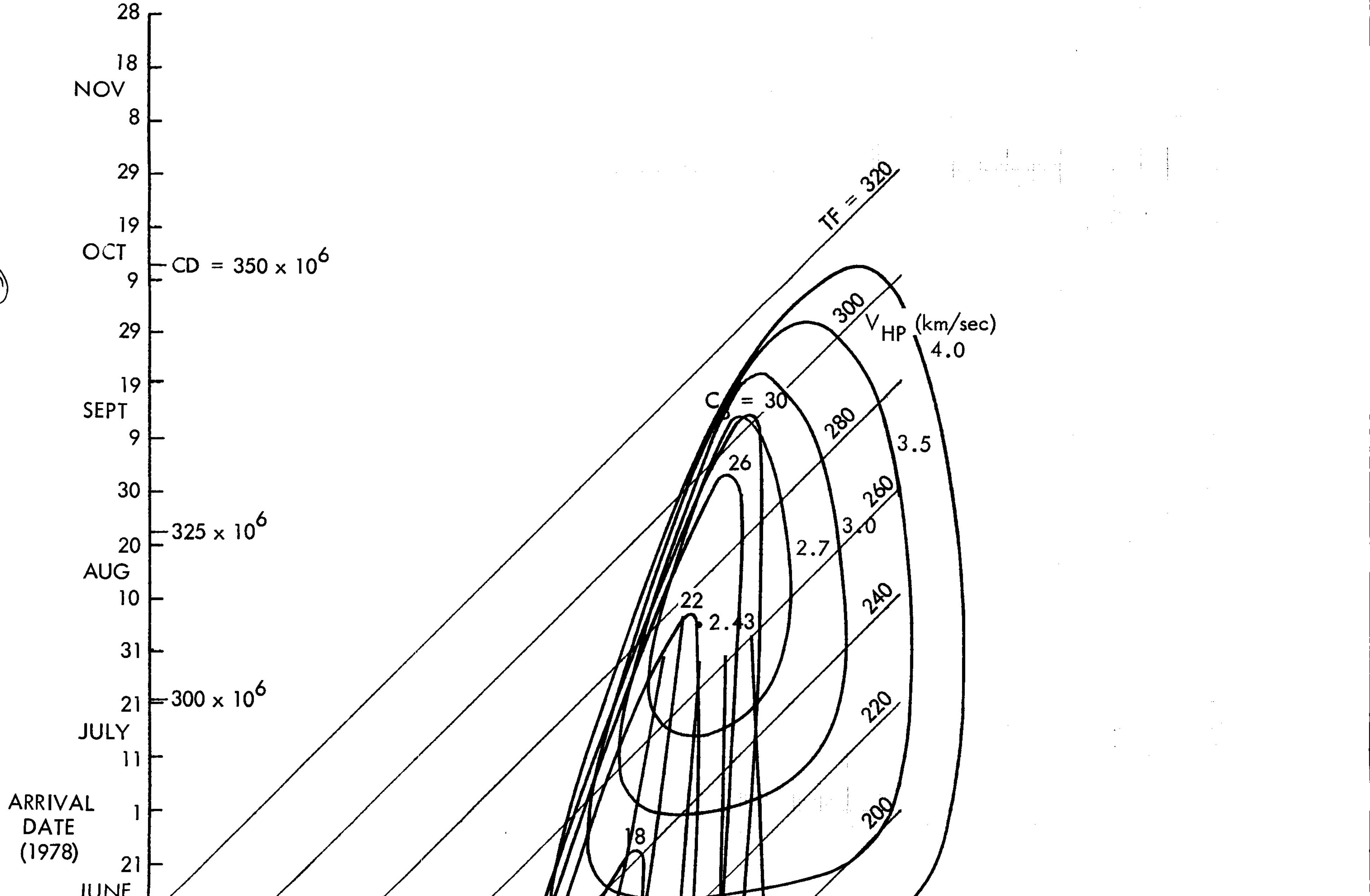


Figure 3.1-40: Earth-Mars Transfer Characteristics — 1975 Type II

2


28
18
NOV
8
29
19
OCT
9
29
19
SEPT
9
30
20
AUG
10
31
21
JULY
11
1
21
JUNE
ARRIVAL
DATE
(1978)
CD = 350 x 10^6
325 x 10^6
300 x 10^6
TF = 320
300
280
260
240
220
200
V HP (km/sec)
4.0
3.5
3.0
2.7
2.43
C3 = 30
26
22
18
75 D

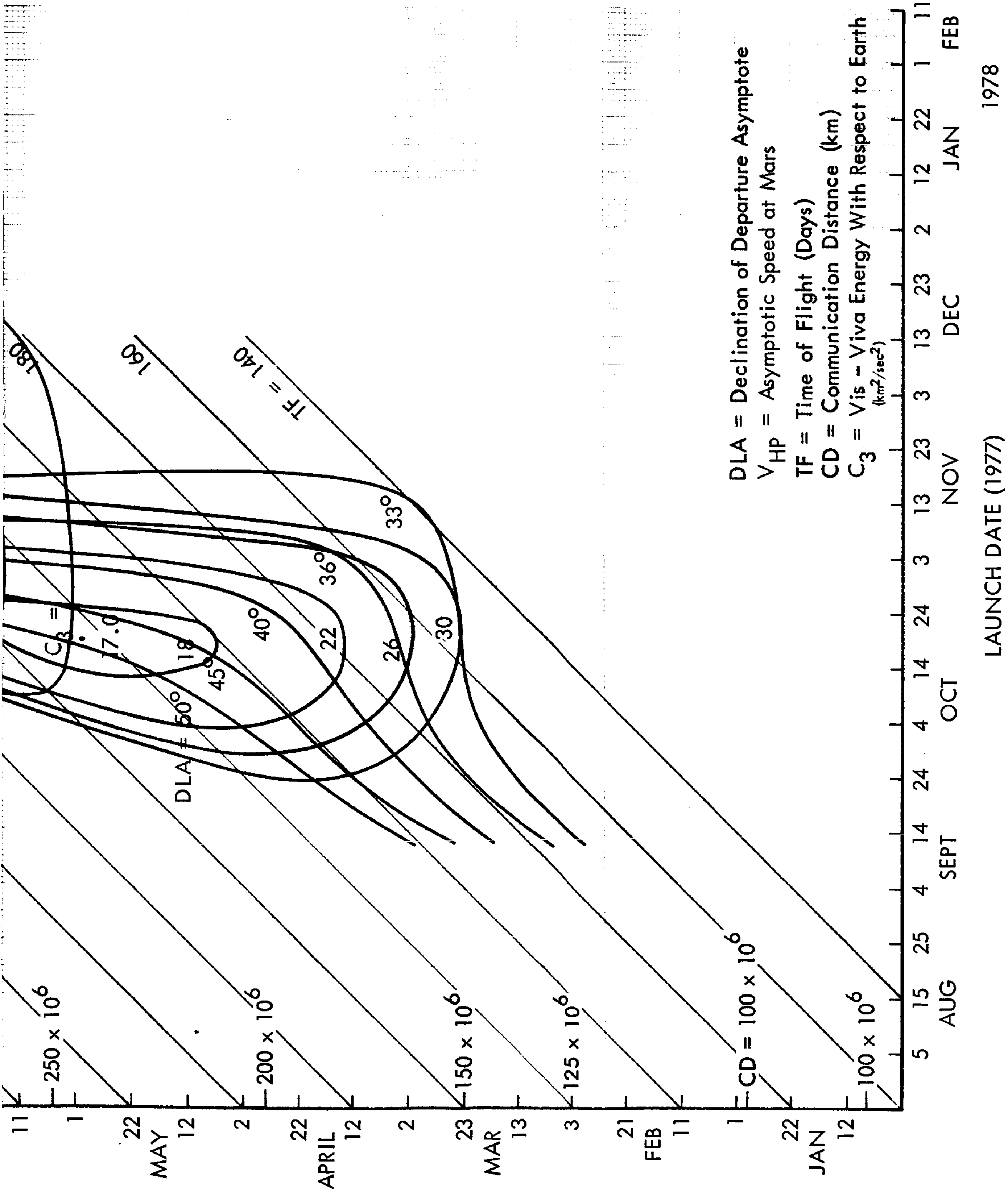
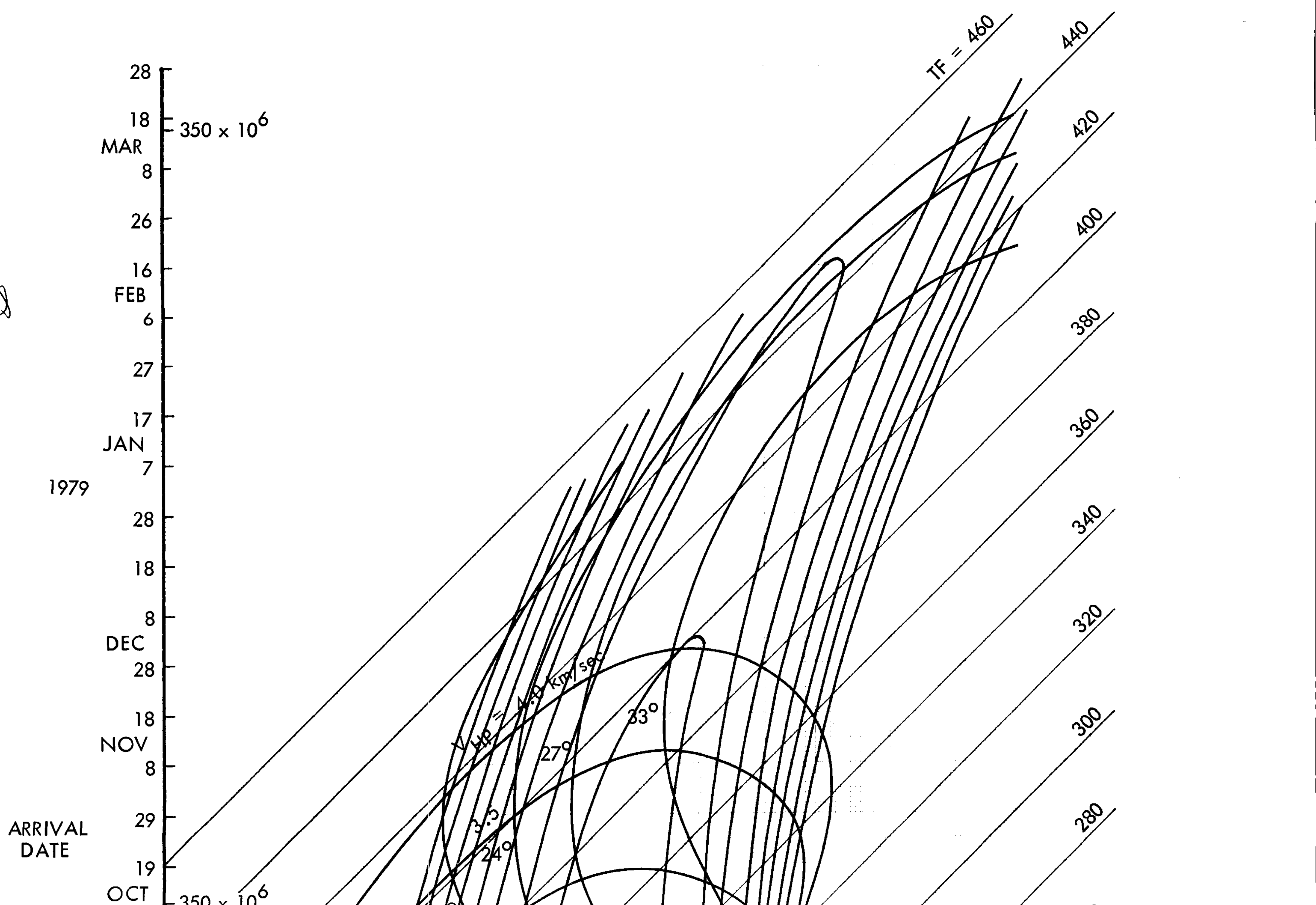


Figure 3.1-41: Earth-Mars Transfer Characteristics — 1977 Type I



ARRIVAL DATE
1979
28
18
MAR
8
26
16
FEB
6
27
17
JAN
7
28
18
8
DEC
28
18
NOV
8
29
19
OCT
350 x 10^6
TF = 460
440
420
400
380
360
340
320
300
280
V_HP = 4.0 km/sec
3.5
33°
27°
24°

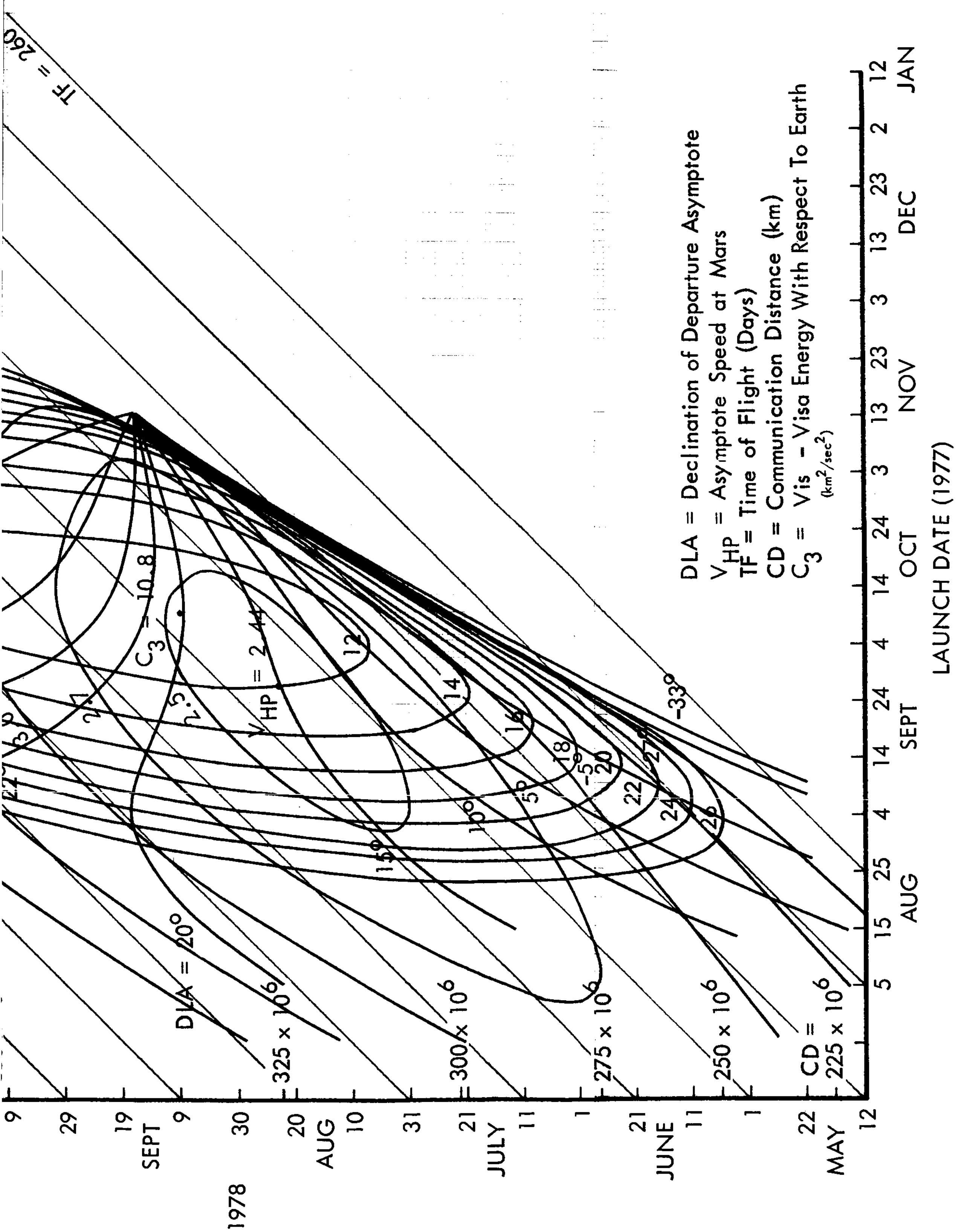
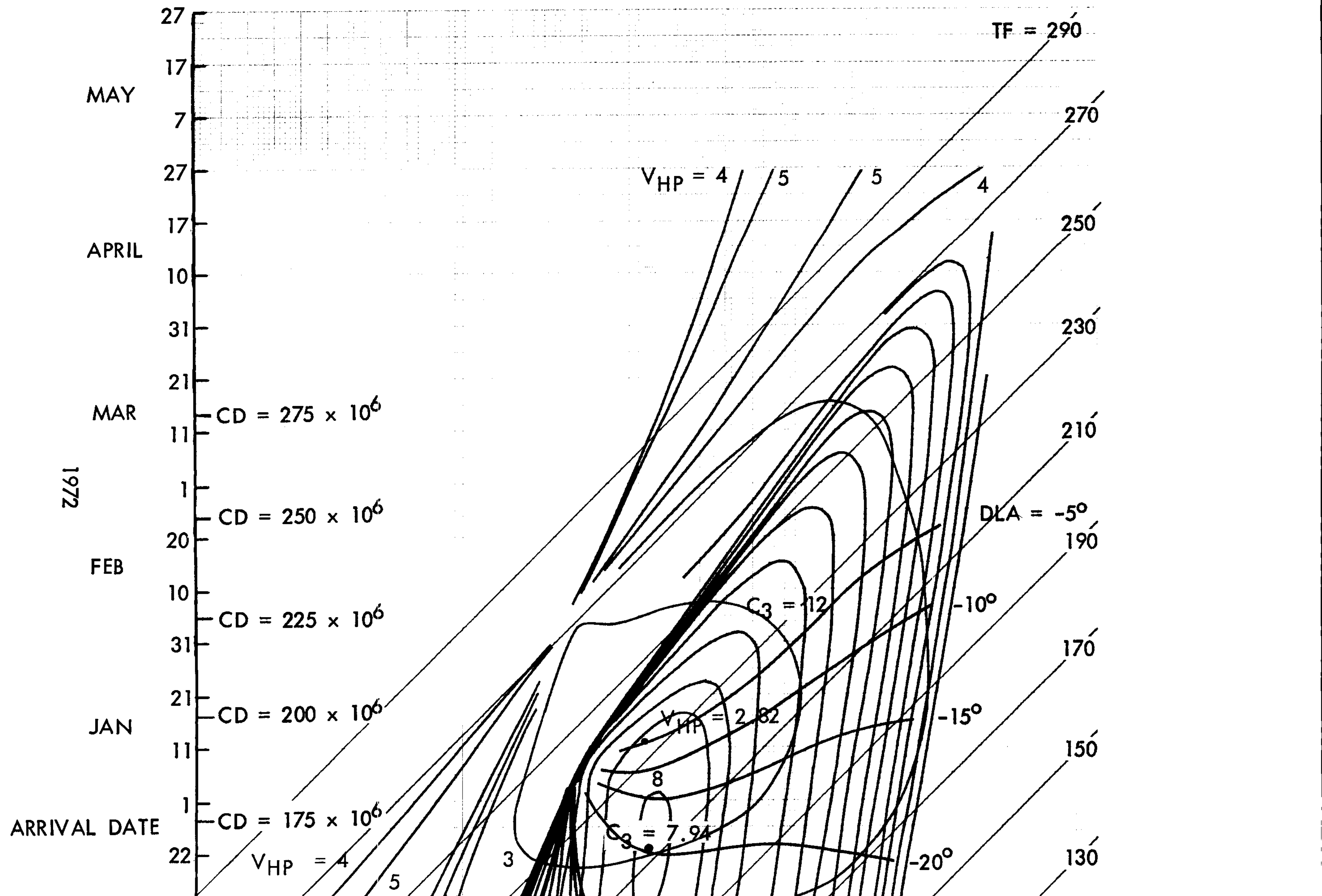


Figure 3.1-42: Earth-Mars Transfer Characteristics — 1977 Type II



TF = 290
270
250
230
210
DLA = -5°
190
170
150
130
-10°
-15°
-20°
V_{HP} = 4
5
5
4
C_3 = 12
V_{HP} = 2.82
8
C_3 = 7.94
3
V_{HP} = 4
5
CD = 275 × 10^6
CD = 250 × 10^6
CD = 225 × 10^6
CD = 200 × 10^6
CD = 175 × 10^6
27
17
7
27
17
10
31
21
11
1
20
10
31
21
11
1
22
MAY
APRIL
MAR
FEB
JAN
1972
ARRIVAL DATE

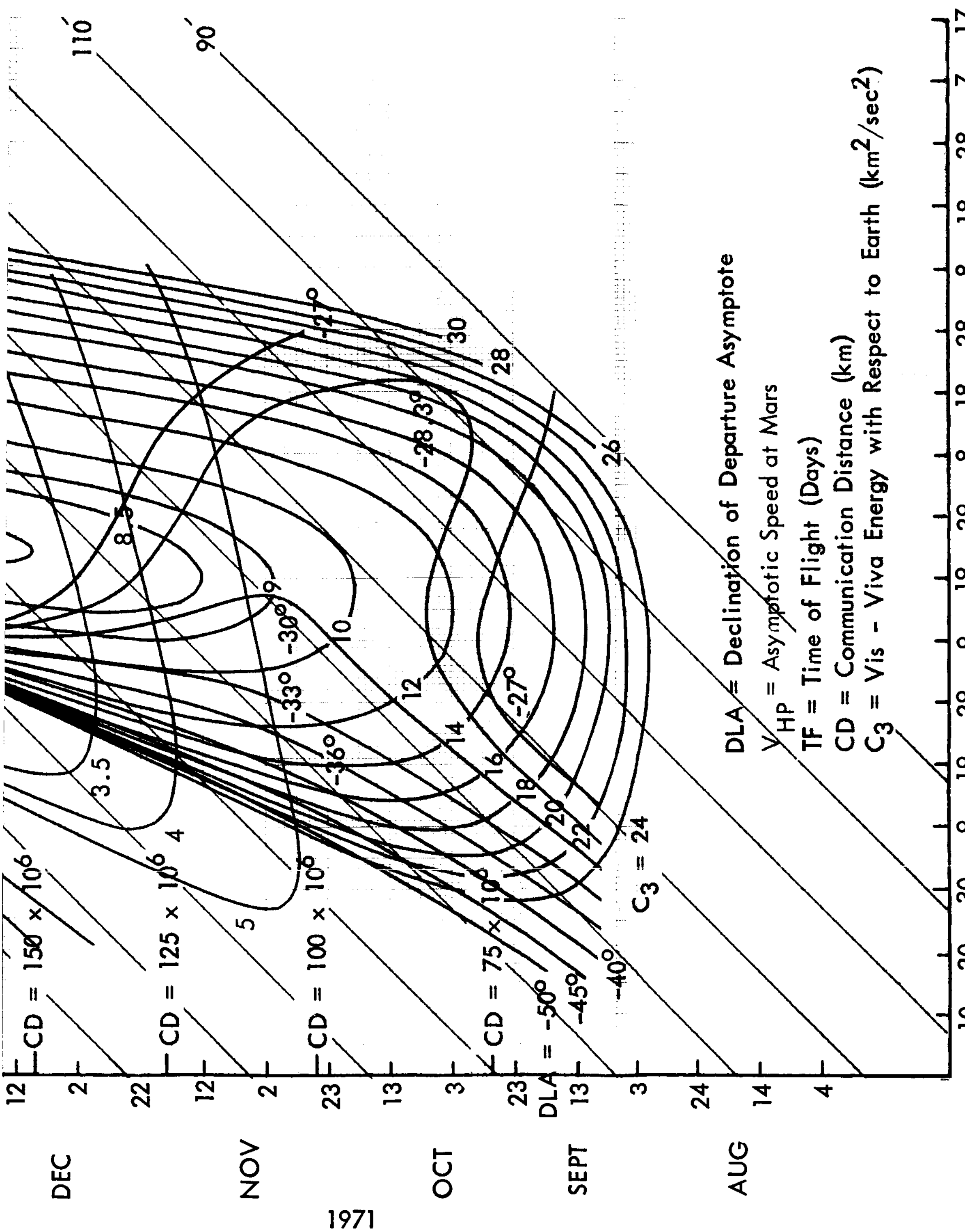


Figure 3.1-43: Trajectory Design Chart, 1971 Type I

3.2 VOYAGER ORBIT-DETERMINATION CAPABILITY

3.2.1 Scope

The results of the orbit-determination accuracy analysis are presented in this section for the trans-Mars, Mars-approach, and Mars orbit mission phases.

3.2.2 Description

The trans-Mars trajectory examined is a Type I, with launch on May 14, 1971, and arrival at Mars on December 15, 1971 (a 215 day transit time). The nominal orbit about Mars has an eccentricity of 0.635, a period of 18 hours, periapsis radius of 6060 kilometers, inclination to Mars equator of 34°, and inclination to plane normal to the Earth-Mars line of 78°. The details of the analysis are presented in Volume B, Section 3.1. The orbit-determination capability during the mission depends on the orbit parameter values and on the sequence of orbit correction maneuvers. The results presented are for specific nominal orbits.

3.2.2.1 Midcourse and Mars-Approach Orbit Determination

The orbit-determination accuracy during the midcourse, or trans-Mars, and Mars-approach mission phases is summarized in Table 3.2-1 for the nominal maneuver timing sequence, 5, 25, and 175 days from launch.

3.2.2.2 Mars Orbit Determination

The dominant error in the initial determination of the orbit about Mars is in the position of the node in the plane normal to the Earth-Mars line. With DSN, it is determined to 0.15 degrees in 2 days and 0.03 degrees

TABLE 3.2-1

ORBIT-DETERMINATION UNCERTAINTIES

Time (Days From Launch)	3σ RSS Position Uncertainty (km)	3σ RSS Velocity Uncertainty (m/sec)
	RELATIVE TO EARTH	
2	4	0.008
5	1.6	0.004
25	10	0.01
50	12	0.004
75	11.5	0.003
100	11	0.002
125	10.8	0.002
150	7	0.002
175	6	0.003
200	56	0.01
205	52	0.01
210	46	0.045
212	42	0.01
214	28	0.02
	RELATIVE TO MARS	
Time to Encounter (Hours)	3σ Error in Estimating Periapsis Radius (km)	
20	300	
10	270	
5	120	

in 3 days. The position error due to a node error of 0.03 degrees is 3 kilometers at apoapsis.

The errors in determination of all other orbit parameters result in position errors of 0.5 kilometer or less in one orbital period using DSN. The dominant source of error is the accumulated effect of system parameters that are omitted or imperfectly represented in the system model.

3.3 VOYAGER FLIGHT SPACECRAFT COMPONENTS DESIGN PARAMETERS

The following paragraphs define the terminology and use of the Spacecraft Components Design Parameters Sheets (SCDPS) in Table 3.3-1.

3.3.1 Scope

3.3.1.1 The SCDPS will serve as the controlling specification for the maximum weight, power, volume, and thermal operating ranges, as well as reliability of all the Model 945-6026 spacecraft subassembly components contained in the following subsystems:

Spacecraft Telecommunications

Attitude Reference Subsystem

Autopilot Subsystem

Reaction Control Subsystem

Central Computer and Sequenceer Subsystem

Electrical Power Subsystem

Spacecraft Structure Subsystem

Spacecraft Mechanisms Subsystem

Temperature Control Subsystem

Pyrotechnic Subsystem

Installation Cables and Tubing

Midcourse Correction Propulsion Subsystem

Orbit Insertion Propulsion Subsystem

Science Payload Instrumentation (GFE)

Science Payload Data Automation System (GFE)

3.3.2 Applicable Document

MC-4-120-E - Functional Specification, Mariner C Flight Equipment (Spacecraft Components Design Parameters).

3.3.3 Subsystem Reference Numbers

Subsystem reference numbers are assigned to various spacecraft subsystems for identification. The numbers are identical to those shown on the equipment drawing tree in Figure 3.4-1.

3.3.4 Design Parameters

The SCDPS lists the number required, weight, volume, power, thermal conditions, reliability, make or buy of all appliable spacecraft components, and the status as to whether the parts within a subassembly or component are on the Voyager approved parts list.

3.3.5 Explanation of the Symbols and Nomenclature Used on the SCDPS

3.3.5.1 Assembly and Subassembly--Column 1

Items listed are subassemblies and components. The subassembly and component names appearing in this column will appear in all applicable documents, equipment lists, and drawing trees.

3.3.5.2 Number Required--Column 2

The number specified in this column defines the quantity required per spacecraft.

				0.1 Sec																	
Pressurization System--Plumbing		1.2.2.1.4			(16.0)									40	120						
Squib Valve, NC	4	1.2.2.1.4.1			2.0																
Squib Valve, NC, Dual	1	1.2.2.1.4.2			0.8				25	6											
Squib Valve, NO	4	1.2.2.1.4.3			3.2																
Solenoid Valve, NO	1	1.2.2.1.4.4			0.8																
Filter	3	1.2.2.1.4.5			1.0																
Relief Valve, Pressure	1	1.2.2.1.4.6			0.5																
Regulator	2	1.2.2.1.4.7			3.0			6 25 Watt 0.1 Sec. One Shot													
Valve & Cap, Fill & Test	3	1.2.2.1.4.8			0.4																
Lines, Connectors & Clamps	-	1.2.2.1.4.9			4.3			7 Included in Reaction Control Subsystem						40	120						
Pressurization System Gas Supply					7																
SUBSYSTEM TOTALS					508.4																
ORBIT INSERTION PROPULSION	1	1.2.2.2		2686	2686									40			30	90	B	.9995	
Rocket Engine, Solid-Inserts	1	1.2.2.2.1			(272.0)											N/A					
Motor Case	1	1.2.2.2.1.1			71.0		40,000								130						
Motor Internal Insulation, Unexpendable	1	1.2.2.2.1.2			32.0										5800						
Motor Internal Insulation Expendable	1	1.2.2.2.1.3			45.0										5800						
Motor Liner	1	1.2.2.2.1.4			3.0										5800						
Motor Nozzle, Unexpendable	1	1.2.2.2.1.5			85.0																
Motor Nozzle, Expendable	1	1.2.2.2.1.6			20.0																
Motor Igniter and S & A	1	1.2.2.2.1.7			4.0										90						
Motor Igniter Prop.	1	1.2.2.2.1.8			2.0										90						
Motor Skirt	1	1.2.2.2.1.9			8.0										220						
Motor Fastener, Seals, etc.		1.2.2.2.1.10			2.0										220						
Thrust Vector Control Unit	1	1.2.2.2.2			(108)									40			30	90			
Torus Tank Welded Assy.	1	1.2.2.2.2.1			23.0										130	90					
Bladder Installation	1	1.2.2.2.2.2			2.0										130	90					
Pressure Plate Assembly	1	1.2.2.2.2.3			1.0										130	90					
Injector Valve Units	4	1.2.2.2.2.4			10.0										130	90					
Servo Valves	4	1.2.2.2.2.5			2.0										130	90					
Burst Discs	4	1.2.2.2.2.6			0.8										90	50					
Filters, Freon	4	1.2.2.2.2.7			0.4										90	50					
Clamps, Injector-to-Tank	4	1.2.2.2.2.8			0.8										90	50					
Hydraulic Installation	1	1.2.2.2.2.9			4.7										90	50					
Valve and Cap Assembly, Freon, Fill and Test	1	1.2.2.2.2.10			0.3										90	50					
Valve, N_2, Squib Oper., NC, Dual	1	1.2.2.2.2.11			0.8										90	50					
Valve N_2, Squib Oper., NO, Dual	1	1.2.2.2.2.12			0.8										90	50					
Lines, Connector, and Clamps - N_2		1.2.2.2.2.13			0.4										90	50					
Freon-Residual					5.0																
Freon - Usable		1.2.2.2.2.14			56.0										90	50					
Solid Propellant		1.2.2.2.3			(2306.0)										90	50					
SUBSYSTEM TOTALS					2686.0																
SCIENCE PAYLOAD INSTRUMENTATION (GFE)		1.2.3.1		195	195									14	158	10.0	-94	275	GFE	.6726	
TV Camera Assemblies (2)	2	1.2.3.1.1												14	131	3.6	+22	176			
Mars Scanner	1	1.2.3.1.2												-40	-4	3.6	-58	158			
Infrared Spectrometer	1	1.2.3.1.3												-110	50	3.6	-110	140			
Dual Frequency Radio Experiment	1	1.2.3.1.4												14	131	3.6	-58	176			
Ultraviolet Spectrometer	1	1.2.3.1.5						100						14	104	3.6	-22	140			
Mars RF Noise Detector	1	1.2.3.1.6												14	131	3.6	-58	176			
Helium Vector Magnetometer	1	1.2.3.1.7												14	131	3.6	-58	176			
Plasma Probe Instrument	1	1.2.3.1.8												14	131	3.6	-58	176			
Trapped Radiation Detector	1	1.2.3.1.9												14	131	3.6	-22	140			
Micrometeroid Detector	1	1.2.3.1.10												14	131	3.6	-94	275			
Ionization Chamber Instrument	1	1.2.3.1.11												-58	193	3.6	-58	508			
SUBSYSTEM TOTALS					195.0																
SCIENCE PAYLOAD DATA Automation System (GFE)		1.2.3.2		55	55									14	131	3.6	-58	176			
Data Automation Equipment		1.2.3.2.1												14	131	3.6	-58	176			
SUBSYSTEM TOTALS					55.0																

SUBSYSTEM TOTALS					[illegible]																
TEMPERATURE CONTROL SUBSYSTEM	1	1.2.1.9		107	107.3			23	91				Unreg. D.C.						M	.9960	
Thermal Control--Propulsion Instl		1.2.1.9.1			(71.4)																
Heat Shield--Solid-Motor Nozzle and Bus Forward-Base Area		1.2.1.9.1.1			32.1									-282	1000	600					
Insulation Blanket--Solid Motor Case & Freon System	1	1.2.1.9.1.2			8.3									-200	300	N.A.					
Heat Shield--Midcourse Rocket Eng.	4	1.2.1.9.1.3			20.4									-300	250	20					
Thermal Shroud--Aft Side Area	1	1.2.1.9.1.4			3.8									-300	250	20					
Thermal Shroud--Aft Base Area	1	1.2.1.9.1.5			4.4									-300	250	20					
Thermal Shroud--Forward-Base Area, Jettisonable	1	1.2.1.9.1.6			1.4									-200	200	NOM					
Louvers--Propulsion Module		1.2.1.9.1.7			1.0									-300	300	N.A.					
Thermal Control (Bus)		1.2.1.9.2			(35.9)																
Louvers--Body Mounted	12	1.2.1.9.2.1			17.5									-300	300	N.A.					
Louvers--Scan Platform	1	1.2.1.9.2.2			3.5									-300	300	N.A.					
Insulation--Body		1.2.1.9.2.3			3.8									-300	250	20					
Insulation--Scan Platform		1.2.1.9.2.4			0.8									-300	250	20					
Thermal Coatings		1.2.1.9.2.5			2.3									-300	250	20					
Temp Sensors	41	1.2.1.9.2.6			1.0									-300	300	N/C					
Solar Shielding	1	1.2.1.9.2.7			2.0									-300	300	20					
Heaters & Wiring	27	1.2.1.9.2.8			5.0									-300	300	N.A.					
SUBSYSTEM TOTALS					107.3																
PYROTECHNIC SUBSYSTEM	1	1.2.1.10			▷3		▷3	2.70	2520		1.0	50 Milsec	O/C Battery	17	120	68			M		
					▷3	NOTE: The weight, volume and reliability of this subsystem are included in the Central Computer and Sequencer Subsystem and in the pyrotechnic operated subsystem components															
INSTL CABLES & TUBING		1.2.1.11		110	110.0															.9999	
Subsystem Cabling/Harness Instl		1.2.1.11.1												-200	250	N/C					
Spacecraft Bus Subsystem Cabling/Harness Inst.		1.2.1.11.2			100									-200	250	N/C					
Spacecraft Propulsion Subsystem Cabling/Harness Inst.		1.2.1.11.3			10									-200	250	N/C					
SUBSYSTEM TOTALS					110.0																
MIDCOURSE CORRECTION PROPULSION	1	1.2.2.1		508	508.4									See Figure 4.3-9 Vol A					B	.9973	
Rocket-Engine System	1	1.2.2.1.1			(22.4)																
Rocket Engine	4	1.2.2.1.1.1			10.0									40	1800		40	120			
Jet Vane Assembly	4	1.2.2.1.1.2			10.4				24					-65	+310		-65	+165			
Pressure Transducers	4	1.2.2.1.1.3			2.0									-65	+165		-65	+165			
Propellant System--Inerts	1	1.2.2.1.2			(75)																
Spherical Tank Welded Assembly	2	1.2.2.1.2.1			26.0		13,050														
Bladder Installation	2	1.2.2.1.2.2			4.0																
Standpipe Installation	2	1.2.2.1.2.3			6.0																
Squib Valve, Normal Closed (NC)	3	1.2.2.1.2.4			2.7									40	120						
Squib Valve, NC, Dual	1	1.2.2.1.2.5			1.5									40	120						
Solenoid Valve, NC	4	1.2.2.1.2.6			1.8				16	▷4											
Squib Valve, Normally Open (NO)	3	1.2.2.1.2.7			2.7																
Squib Valve, NO, Dual	1	1.2.2.1.2.8			1.5																
Solenoid Valve, NO	1	1.2.2.1.2.9			0.9				25	▷5											
Filter	2	1.2.2.1.2.10			1.6																
Relief Valve, Thermal	1	1.2.2.1.2.11			0.4																
Valve & Cap, Fill & Test	3	1.2.2.1.2.12			0.9																
Feed Lines, Connectors, & Clamps		1.2.2.1.2.13			9.4																
Pressure Transducer	1	1.2.2.1.2.14			0.5																
Temperature Transducer	2	1.2.2.1.2.15			0.1																
Residual Hydrazine--Allowances		1.2.2.1.2.16			15.0																
Exp. Hydrazine-Usable		1.2.2.1.3			(395.0)									40	120						

▷4 Duty Cycle 400 Sec. Max

▷5 20-150 Sec. Per Shot

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Panel Deployed Latch	6	1.2.1.6.1.9			8.2			Bridge				millisec.									
Photovoltaic Cell Assy	48,708	1.2.1.6.1.10			92.0			None													
Battery Instl. Ea. 8 x 16 x 6 in.		1.2.1.6.2			(123.0)		2100/T														
Power Conditioning Equipment		1.2.1.6.3			(50.0)			120/Ea.		3/Ea.	26/Ea.			40	110		40	77			
Battery Charger & Failure Sensor	3	1.2.1.6.3.1			9.0		108/Ea.			6.3/Ea.			Unreg. Bus	14	167		14	167			
Booster Converter & Share Sensor	1	1.2.1.6.3.2			1.5		72				7		Jnreg. Bus	14	167		14	167			
DC-DC Regulator	4	1.2.1.6.3.3			25.6		144/Ea.						Jnreg. Bus	14	167		14	167			
Power Switch & Logic	1	1.2.1.6.3.4			3.2		108	628	1.141	11	33		Unreg Bus	14	167		14	167			
Synchronizer	1	1.2.1.6.3.5			1.0		36	1.5	2.0	1.0	1.5		Clock Pulse Reg. Bus	14	167		14	167			
2.4-kc, 1 ϕ Inverter	2	1.2.1.6.3.7			4.0		72/Ea.	34.0/Ea.	83.67	4.0/Ea	8.67/Ea.		Reg. Bus Unreg. Bus	14	167		14	167			
AC Failure Sense	1	1.2.1.6.3.8			1.7		108	NA	NA	1.0	1.5		Reg. Bus	14	167		14	167			
DC Failure Sense	2	1.2.1.6.3.9	65-688		4.0		108/Ea.	NA	NA	1.0/Ea	1.5/Ea.		Unreg. Bus Reg. Bus	14	167		14	167			
SUBSYSTEM TOTALS					457.0																
SPACECRAFT STRUCTURE SUBSYSTEM		1.2.1.7		498	(497.8)														M	.9999	
Equipment Support Structure		1.2.1.7.1			373.2																
Cylindrical shell		1.2.1.7.1.1			(68.7)									-50	200	10					
Vertical truss		1.2.1.7.1.2			(20.2)									-50	200						
Upper support ring		1.2.1.7.1.3			(15.3)									-50	200						
Lower support ring		1.2.1.7.1.4			(46.1)									-50	200						
Lower support truss		1.2.1.7.1.5			(101.3)									-50	200	10					
Secondary support structure		1.2.1.7.1.6			(121.6)									-300	200	10					
High-gain antenna supports		1.2.1.1.7.1.6.1			2.0																
Low-gain antenna supports		1.2.1.7.1.6.2			5.9																
VHF antenna supports		1.2.1.7.1.6.3			5.9																
Science boom		1.2.1.7.1.6.4			23.1																
Science boom supports		1.2.1.7.1.6.5			11.7																
Scan platform		1.2.1.7.1.6.6			24.0																
Scan platform supports		1.2.1.7.1.6.7			8.7																
Solar panel attach structure		1.2.1.7.1.6.8			8.0									-300	200	10					
Equipment attach structure		1.2.1.7.1.6.9			12.0									-50	200	10					
Meteoroid shielding		1.2.1.7.1.6.10			20.3									-300	200	10					
Propulsion/Reaction Control Support Structure		1.2.1.7.2			124.6																
Motor support assy		1.2.1.7.2.1			30.5									-50	200	10					
Tank support assy		1.2.1.7.2.2			33.1																
Engine support assy		1.2.1.7.2.3			7.2																
Misc supports & fasteners		1.2.1.7.2.4			4.8									-50	200	10					
Meteoroid shielding		1.2.1.7.2.5			49.0									-300	200						
SUBSYSTEM TOTALS					497.0																
SPACECRAFT MECHANISMS SUBSYSTEM		1.2.1.8		59	58.9														M	.9988	
High-gain Antenna Mech Instl		1.2.1.8.1			(31.5)									-70	167	20					
Boost supports		1.2.1.8.1.1			1.4																
Deployment linkage		1.2.1.8.1.2			19.9																
Deploy pin pullers		1.2.1.8.1.3			0.4							2									
Dish pin pullers		1.2.1.8.1.4			0.3							2									
Pointing mechanisms		1.2.1.8.1.5			9.5																
Low-gain Antenna Mech Instl		1.2.1.8.2			(2.9)									-70	167	20					
Deploy pin puller		1.2.1.8.2.1			0.1							2									
Deploy Mechansims		1.2.1.8.2.2			2.5																
Locking Mechansims		1.2.1.8.2.3			0.3																
VHF Antenna Mechanism Instl		1.2.1.8.3			(4.4)									-70	167	20					
Deploy pin puller		1.2.1.8.3.1			0.2							2									
Locking Mechanisms		1.2.1.8.3.3			4.2																
Scan Platform Mechanism Instl		1.2.1.8.4			(11.5)									-70	167	20					
Boost supports		1.2.1.8.4.1			1.8																
Pointing mechanism		1.2.1.8.4.2			8.9																
Cover actuation mechanism		1.2.1.8.4.3			0.8																
Science Boom Mechanism Instl		1.2.1.8.5			(5.4)									-70	167	20					
Deploy pin pullers		1.2.1.8.5.1			0.2																
Deploy mechansim		1.2.1.8.5.2			4.0																
Locking mechanisms		1.2.1.8.5.3			1.2																
Bacteriological Barrier Mechanism Instl		1.2.1.8.6			(3.2)									-70	167	20					
Pin pullers		1.2.1.8.6.1			0.4							2									
Separation		1.2.1.8.6.2			2.8																

2 NOTE: Pin puller power is supplied from pyrotechnic subsystem

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Internal Cabling		1.2.1.1.4.4			1.0														
SUBSYSTEM TOTALS					207.0														
ATTITUDE REFERENCE SUBSYSTEM	1	1.2.1.2		51	50.8	3872	33.5	33.5	33.5	33.5	100%	1ø 2400 +35vd.c						B	.9969
Inertial Reference Unit	1	1.2.1.2.1			(22.4)		29.3	31.8	29.3	31.8		35 vd.c	-65	165		-80	165		
Gyro (G10)	3	1.2.1.2.1.1			8.0		29.3	31.8	29.3	31.8		Plus 24 kc	178	182					
Gyro Electronics	1	1.2.1.2.1.2			9.9		.3	.5	.3	.5									
Accelerometer	2	1.2.1.2.1.3			4.5		1	1	1	1			-60	200					
													-150	300					
Sun Sensor - Primary	2	1.2.1.2.2			(2.5)		.1	.1	.1	.1	Cont.	35 v.dc	+3	149	30	-20	160		
Secondary													-10	185					
Canopus Sensor	2	1.2.1.2.3			(13.5)		5	5	5	5	Cont.	35 vd.c	0	104	-	-30	125		
Attitude Ref. Subs Packaging					(10.9)														
Remote Sun Sensors	3	1.2.1.2.4			(1.5)	80													
1-1/2 Hours (Acquisition)																			
Several Minutes During Injection																			
SUBSYSTEM TOTALS					50.8														
AUTOPILOT SUBSYSTEM	1	1.2.1.3		11	11.0	640	7	79	7.5	15.0	10 min.	35 v dc	35	95			167	B	.9998
Summing Amplifier	3	1.2.1.3.1																	
Derived Rate Networks	3	1.2.1.3.2																	
Valve Drivers (RC)	16	1.2.1.3.3																	
Driver Amplifiers (JV)	16	1.2.1.3.4																	
Valve Drivers (SI)	4	1.2.1.3.5																	
Control Logic and Switching	1	1.2.1.3.6																	
Power Supply	1	1.2.1.3.7																	
SUBSYSTEM TOTALS					11.0														
REACTION CONTROL SUBSYSTEM		1.2.1.4.1		212	(212)													M	.9996
N_2 Tanks	4	1.2.1.4.2			128.0	8660							30	100		30	110		
Pressure Regulator	2	1.2.1.4.3			4.0								-65	165		-65	165		
Valve--Sol. Oper. Latching	2	1.2.1.4.4			3.6							C.C.&S	-65	165		-65	165		
Valve & Cap--Fill	2	1.2.1.4.5			0.3								-65	165		-65	165		
Thruster Units	4	1.2.1.4.6			9.6							Autopilot	-65	165		-65	165		
Gas Lines, Connectors, Clamps		1.2.1.4.7			4.0														
Pressure Transducers	2	1.2.1.4.8			1.0														
Temperature Transducers	4	1.2.1.4.9			0.2														
Quad Check Valve	2	1.2.1.4.10			0.6														
Filter	2	1.2.1.4.11			0.7														
N_2 Propellant	-				60								-90	165		-90	165		
SUBSYSTEM TOTALS					212.0														
CENTRAL COMPUTER & SEQUENCER		1.2.1.5		58.0	58													M	.9945
Control Assembly	1	1.2.1.5.1			(29)	1550 8x8x24	35	35	35	35	1.0	1.2.1.6	32	150	-	32	150		
Logic Card (Integrated Circuit) Subassembly	21	1.2.1.5.1.1			13														
Memory Subassembly	2				4														
Oscillator Subassembly	2	1.2.1.5.1.4			1														
Cabling/Wiring		1.2.1.5.1.6			6.0														
Structure		1.2.1.5.1.7			5														
Switching Assembly	1	1.2.1.5.2			(29)	1050 8x8x16	5.5	780	5.5	780	50 ms	4.25	32	150	-	32	150		
							Squib Firing												
Cordwood Module	9	1.2.1.5.2.1	108 Squib Drivers 8 Valve Drivers 15 Relay Drivers	-12															
Relay Subassembly (2 Relays)	1	1.2.1.5.2.2		4															
Structure		1.2.1.5.2.3		5															
Cabling/Wiring		1.2.1.5.2.4		8															
SUBSYSTEM TOTALS					58.0	2600													
ELECTRICAL POWER SUBSYSTEM		1.2.1.6		457	(457.0)													B	.9923
Solar Panel Array Instl.	1	1.2.1.6.1			(284.0)		NA					NA	-275	175					
Solar Panel Frame	3	1.2.1.6.1.2			53.8														
Substrate Assembly	12	1.2.1.6.1.3			69.5														
Fasteners					0.5														
Solar Panel Strut	6	1.2.1.6.1.4			22.5														
Panel Deploy Devices & Hnge.	6	1.2.1.6.1.5			23.6														
DC Motor	6	1.2.1.6.1.6					10/Mot	75/M	7.25/M	75/M	2 sec.	Battery				NA	NA		
Panel Boost Latch	12	1.2.1.6.1.7			13.9														
Squib (Double Bridge)	12	1.2.1.6.1.8					20/	50/B	NA	NA	100	Squib Firing							

Table 3.3-1: Spacecraft Components Design Parameters Sheet (SCDPS)

ASSEMBLY AND SUBASSEMBLY	NUMBER REQD/SPACECRAFT	ITEM NUMBER	DRAWING NUMBER	ALLOCATED WEIGHT (LBS.)	CURRENT EST. WEIGHT (LBS.)	ALLOCATED VOLUME (IN^3)	CURRENT EST. VOLUME (IN^3)	ELECTRICAL POWER DISSIPATION & SOURCES: INPUT WATTS AVE.	INPUT WATTS PEAK	DISSIPATED WATTS AVE.	DISSIPATED WATTS PEAK	PEAK POWER DUTY	PRIMARY POWER SOURCE	FLIGHT APPROVED THERMAL OPERATING CONDITIONS: MIN. TEMP °F.	MAX. TEMP °F.	MAX. TEMP CHANGE ° F/MIN	THERMAL NON-OPERATING CONDITIONS: MIN TEMP °F	MAX. TEMP °F	MAKE OR BUY	CURRENT RELIABILITY ASSESSMENT	ON VOYAGER APPROVED LIST
COLUMN NUMBER 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22
TELECOMMUNICATIONS		1.2.1.1	INFORMATION TO BE ADDED TO THIS COLUMN DURING PHASES IB AND II	207.0	207.0	VOLUME CURRENTLY AVAILABLE AS REQUIRED													B	.8416	INFORMATION TO BE ADDED TO THIS COLUMN DURING PHASES IB AND II
Radio Subsystem		1.2.1.1.1																			
					87.2			37.0		36.0	Launch										
Receiver & Exciter Assy		1.2.1.1.1.1			(36.8)		16x8x8	20.0		20.0	Cruise		+35±1.75vd.c	35	95	2°	-13	+167			
Receivers	2	1.2.1.1.1.1.1			8.5			5.6		5.6											
Command Detector	2	1.2.1.1.1.1.2			6.0			1.5		1.5											
Ranging	2	1.2.1.1.1.1.3			1.0			1.0		1.0		Up to L+20 days									
Exciter	2	1.2.1.1.1.1.4			2.8			5.4		5.2											
Launch Xmitter	1	1.2.1.1.1.1.5			2.2			20.0		19.0		Launch									
Redundancy Control & Logic	1	1.2.1.1.1.1.6			4.5			0.6		0.6											
Preamp	1	1.2.1.1.1.1.7			1.0			0.25		0.25					86			+158			
Convertor-Exciter No. 1	1	1.2.1.1.1.1.8			1.3			1.1		1.1					95			+167			
Conv.-Launch Tx & Ex No. 2	1	1.2.1.1.1.1.9			1.3			1.1		1.1		Cruise									
								4.3		4.3		Launch									
Conv.-Rec./Com./Rang.	2	1.2.1.1.1.1.10			2.7			1.9		1.9											
Conv.-Rec. Control Logic	1	1.2.1.1.1.1.11			1.3			1.7		1.7											
Internal Cabling	-	1.2.1.1.1.1.12			4.2																
RF PA Assembly		1.2.1.1.1.2			(50.4)		16x8x26	0.2		0.2		Launch	+35±1.75vd.c								
								150.2		150.2		Cruise									
RF - Structure	1	1.2.1.1.1.2.1			3.0									35	113	2°	-13	+185			
Diplexer	1	1.2.1.1.1.2.2			1.5																
4-Port Circulator	3	1.2.1.1.1.2.3			8.2			0.18		0.18											
Preselector	2	1.2.1.1.1.2.4			1.5																
PA--Structure	1	1.2.1.1.1.2.5			4.0																
TWTA	2	1.2.1.1.1.2.6			18.0			150		100								+185			
Isolator	3	1.2.1.1.1.2.7			4.6										85			+185			
Band Pass Filter	3	1.2.1.1.1.2.8			3.0																
Band Rejection Filter	1	1.2.1.1.1.2.9			0.3																
Hybrid	1	1.2.1.1.1.2.10			1.0																
Circulator Sw. (3-Port)	1	1.2.1.1.1.2.11			0.8			0.03		0.03											
Internal Cabling	-	1.2.1.1.1.2.12			4.5																
Telemetry Subsystem		1.2.1.1.2			58.0																
Planetary Science Tape Recorder Assembly	1	1.2.1.1.2.1			(32.0)		16x8x28	1.2		1.2		Liftoff	+35°1.75vd.c								
								25		25		Orbital									
Tape Recorders	2	1.2.1.1.2.1.1			30.0			25 Record		25				35	86	2°	-13	+122			
								2.4 Play-		2.4		Orbital									
								1.2 back		1.2		Liftoff									
Internal Cabling					2.0																
Telemetry Processor Assembly		1.2.1.1.2.2			(26.0)		16x8x7	6.0		6.0				35	95	2°	-13	+167			
Engineering Core Memory	1	1.2.1.1.2.2.1			5.0			0.5		0.5		Launch									
												Cruise									
Capsule Core Memory	1	1.2.1.1.2.2.2			5.0			0.5		0.5											
Eng. Multiplexer/Encoder/Mod.	1	1.2.1.1.2.2.3			11.5			3.3		3.3											
Converter	1	1.2.1.1.2.2.4			1.5			1.7		1.7											
Internal Cabling		1.2.1.1.2.2.5			3.0																
Spacecraft Antenna Subsystem	1	1.2.1.1.3			54.3																
Low-Gain Antenna	1	1.2.1.1.3.1			6.4																
High-Gain Antenna	1	1.2.1.1.3.2			38			1.0	120.0	1.0	120.0	Standby									
								6.0	120.0	6.0	120.0	Maneuvers									
Shroud Antenna	1	1.2.1.1.3.3			2.0																
Launch/Acquisition Antenna	1	1.2.1.1.3.4			1.5																
Relay Antenna (VHF)	1	1.2.1.1.3.5			6.4																
Relay Radio Subsystem		1.2.1.1.4			7.5		8x8x9	5.2		5.2		Encounter	+35+1.75vd.c	35	95	2°	-13	+167			
Receiver	1	1.2.1.1.4.1			3.0			2.10		2.10											
T/M Detector	1	1.2.1.1.4.2			2.5			1.53		1.53											
Converter	1	1.2.1.1.4.3			1.0			1.57		1.57											

Table 3.3-2: SUMMARY TOTAL

SUBSYSTEM	Summary Weights Science Payload	S/C Bus	Propul. Inst.	Reliability
Spacecraft Telecommunications		227		0.8416
Attitude Reference Subsystem		51		0.9969
Autopilot Subsystem		11		0.9999
Reaction Control Subsystem		212		0.9996
Central Computer and Sequencer Subsystem		58		0.9945
Electrical Power Subsystem		457		0.9923
Spacecraft Structure Subsystem		373	125	0.9999
Spacecraft Mechanisms Subsystem		59		0.9988
Temperature Control Subsystems		36	71	0.9960
Pyrotechnic Subsystem		-	-	Reliability included in CC&S
Installation Cables		100	10	0.9999
Midcourse Correction Propulsion Subsystem			508	
Orbit Injection Propulsion Subsystem			2686	0.9968
Science Payload Instrumentation	195			SC/PL Planetary exper. only = .6726
Science Payload Data Automation System (GFE)	55			
Contingency		166	*100	
Spacecraft System Total	250	1750	3500	0.552

*Including 38-pound allowance for possible change from fiberglass to Ti solid motor case

3.3.5.3 Item Number--Column 3

This column indicates a specific number assigned to the subassembly and components as established by the drawing tree shown in Figure 3.4-1. This number serves as a positive identification until engineering drawing numbers become available.

3.3.5.4 Drawing Number--Column 4

Numbers shown are the numbers assigned to Engineering drawings, which define the installation, subassembly, or component.

3.3.5.5 Weight--Columns 5 and 6

Weight shown in Column 5 represents allocated weights for the total number of subassemblies or components making up the spacecraft configuration. Allocated weights establish the maximum permissible values. Weights shown in Column 6 represent current weights for the subassemblies and components on the date indicated in the upper right hand corner of the SCDPS.

3.3.5.6 Volumne--Columns 7 and 8

Volumes shown in Column 7 represent allocated volume for subassemblies or components making up the spacecraft configuration. Allocated volume establishes the maximum permissible values.

Weight shown in Column 8 represents current volume for the subassemblies and components on the date indicated in the upper right hand corner of the SCDPS.

3.3.5.7 Electrical Power, Dissipation and Sources--Columns 9 through 14

Input Power: Columns 9 and 10 represent actual input watts, average and peak, respectively, during operation.

Power Dissipation: Column 11 and 12 represent actual dissipated watts, average and peak, respectively, during operation.

Duty Cycle: Column 13 represents peak power duty cycle during operating time of each subassembly and component.

Primary Power Source: Column 14 lists the power source for each subassembly.

3.3.5.8 Thermal Operating Conditions--Columns 15 through 17

Columns 15 and 16 represent the minimum and maximum subassembly or component temperature of degrees Fahrenheit that the subassembly or component is designed to withstand under normal operating conditions.

Column 17 lists the maximum rate of temperature change in degrees Fahrenheit per minute under normal operating conditions.

3.3.5.9 Thermal Nonoperating Conditions--Columns 18 and 19

Columns 18 and 19 represent the minimum and maximum temperature in degrees Fahrenheit that the subassembly or component is required to withstand when in a nonoperating condition.

3.3.5.10 Make or Buy--Column 20

Column 20 defines whether the equipment is purchased or Boeing-made. The Government Furnished Equipment (GFE) associated with the science payload is indicated on Table 3.3-1.

3.3.5.11 Reliability--Column 21

Column 21 defines the numerical reliability value assigned to the subassembly or component.

3.3.5.12 Voyager Approved Parts List--Column 22

A "yes" in this column indicates that the subassembly or component contains parts that appear in the Voyager Approved Parts List (VAPL). A "no" in this column indicates that the subassembly or component contains a part or parts not appearing in the VAPL. Further discussion of the VAPL can be found in Section 5.9.1.6.

3.4 VOYAGER EQUIPMENT ELEMENTS AND DOCUMENTATION IDENTIFICATION

3.4.1 Scope

This subsection describes the Boeing method for identifying Voyager Spacecraft Systems equipment elements and related documentation. Equipment elements consist of hardware items (assemblies, subassemblies, parts, etc.) which, when properly assembled, will constitute the contractual end item; related documentation consists of specifications, drawings, documents, etc., which define the hardware items.

3.4.2 Applicable Documents

NPC 500-1, NASA, "Configuration Management Manual," May 18, 1964.

Document D-4900, Boeing, "Drafting Procedures Manual."

3.4.3 Requirements

The Standard Configuration Identification Numbering System, defined in Exhibit X of NPC 500-1 and interpreted in the Boeing Drafting Procedures Manual, will be employed by the Boeing Voyager program. This system recognizes the following classifications:

1) Contract end item (CEI) numbers;
2) Specification identification numbers;
3) Drawing and part numbers;
4) Change identification numbers;

5) Serial numbers;

6) Code identification numbers.

NPC 500-1, Exhibit X specifies that all identifying numbers shall be assigned and controlled by the contractor in accordance with certain minimum standards defined in NPC 500-1; the numbering system reflected in subsequent paragraphs has been designed to fulfill these requirements.

The identifying numbers will be affixed on the equipment elements and related documentation in accordance with the Boeing "Drafting Procedures Manual." Identification number formats will be as described in the following paragraphs.

3.4.3.1 CEI Numbers

The CEI number is a permanent Boeing number assigned to identify all units comprising the contract end-item family (type, model, series), and is the basic (root) number reflected in the CEI detail specification number, the CEI top drawing number, and the CEI top part number.

The CEI number is composed of seven digits as shown in the following example.

A50034A

Suffix Code -- added by the customer and denoting baseline approval status

Root Number -- A50000 through A59999 assigned to the Voyager program

3.4.3.2 Specification Identification Number

The specification identification number is composed of 12 digits as shown in the following example.

CP22A50034A1

Source Code -- used only when two or more sources are available

Suffix Code -- added by the customer and denoting specification approval status

Drawing Number -- note relationship to CEI number

Specification Prefix -- as defined in NCP 50C-1, Exhibit X

3.4.3.3 Top Drawing and Part Numbers

The top drawing number is composed of eight digits as shown in the following example.

22A50034

Root number -- A50000 through A59999 assigned to the Voyager program (note relationship to CEI number)

Drawing Size Code

Boeing Division Code

The top drawing part numbers are composed of 15 digits as shown in the following example.

22A50034-101-11

- 11 — Configuration Number--assigned sequentially starting with 11.
- 101 — Family Designation Code--assigned sequentially starting with 101.
- 22A50034 — Basic Drawing Number

3.4.3.4 Subordinate Drawing and Part Numbers

Subordinate drawing and part numbers are constructed as shown in section 3.4.3.3, except that the drawing root number will not be significant to the CEI number. A subordinate drawing (and part number) is shown in the following example.

25A51234-101-11

- 101-11 — Part Family Designation Code and Configuration Number (see section 3.4.3.3).
- 25A51234 — Basic Drawing Number (see section 3.4.3.3)

3.4.3.5 Change Identification Numbers

Change identification numbers will be assigned to all Class I and Class II change packages. These change packages are defined in ANA Bulletin 445 and referenced in NPC 500-1. Class I changes will be identified by engineering change proposal (ECP) numbers assigned sequentially; Class II changes will be identified by production revision request (PRR) numbers assigned sequentially. Both change classification numbers shall be identified on the applicable drawings.

3.4.3.6 Serial Numbers

All drawings will specify serial numbering requirements when applicable. Serial numbers shall be issued for:

1) All contract end items (CEI's)'

2) All critical components of CEI's;

3) Certain items of equipment below the CEI level, provided that these items are time/cycle sensitive, or that component operational data notices have been issued against these items.

Each of the above categories will have sequentially assigned, seven-digit serial numbers starting with 0000001. Once issued, the numbers will not be used again.

3.4.3.7 Code Identification Numbers

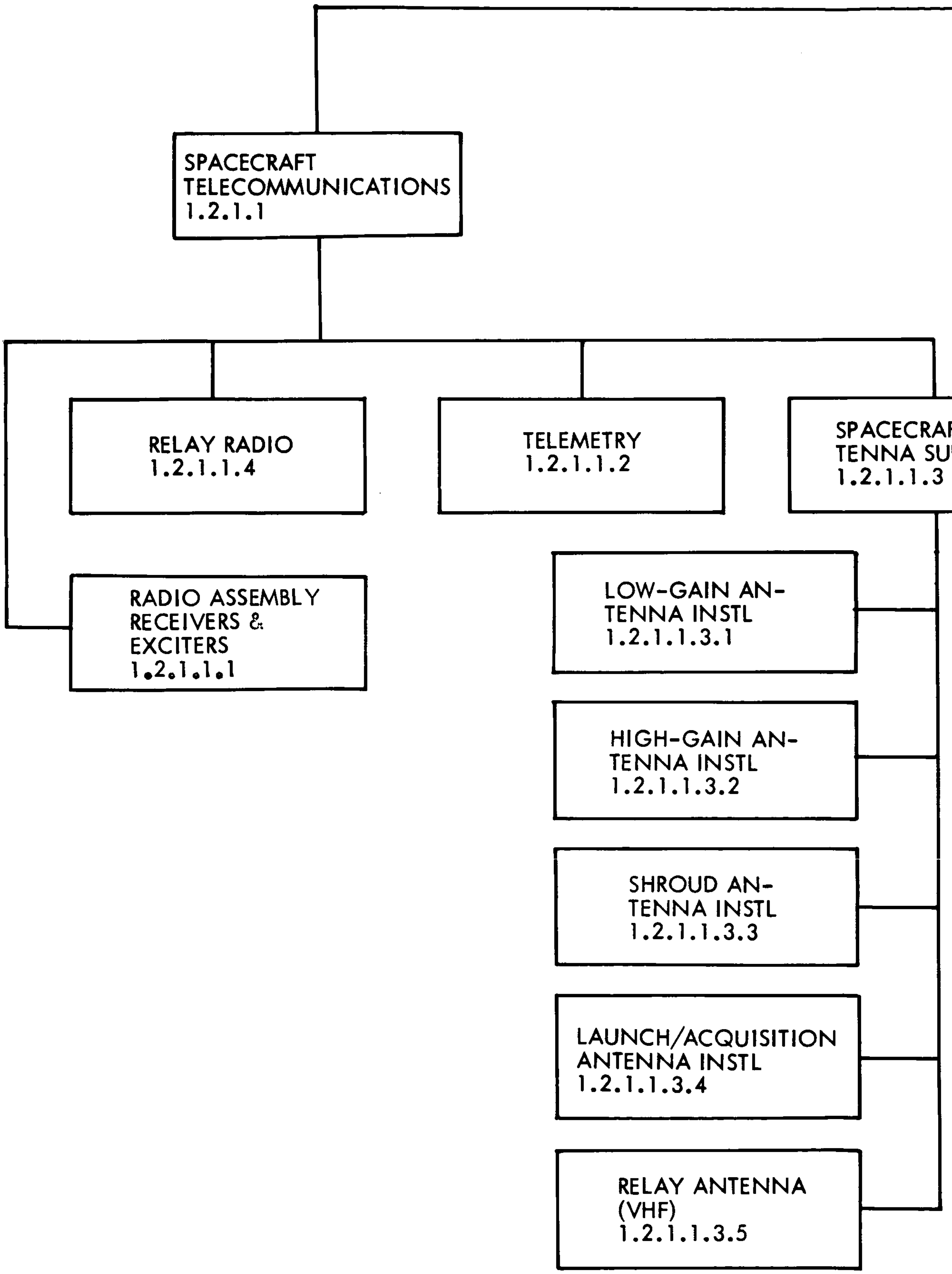
All drawings will be identified to the Boeing code identification number (81205), listed in the Federal Supply Codes for Manufacturers Catalog, H4-1.

3.4.3.8 Equipment/Drawing Tree

The equipment/drawing tree shown in Figure 3.4-1 (Drawing 25-50034, Sheet 4 of 5) depicts engineering subsystem designs (drawings) as related to the Voyager Spacecraft Bus, propulsion subsystem,and Science Payload.

The subsystems are identified by four-digit numbers that are identical to the item numbers assigned to the subsystems by both the Voyager program breakdown structure at the fourth level, and the spacecraft components design parameters in Table 3.3-1.

Once engineering drawing numbers become available for the subsystem described above, the equipment/drawing tree will be revised to incorporate these numbers and include updated subcontractor data and further refinements.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SPACECRAF
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1.2.1.1.3
RADIO ASSEMBLY RECEIVERS & EXCITERS 1.2.1.1.1
LOW-GAIN ANTENNA INSTL 1.2.1.1.3.1
HIGH-GAIN ANTENNA INSTL 1.2.1.1.3.2
SHROUD ANTENNA INSTL 1.2.1.1.3.3
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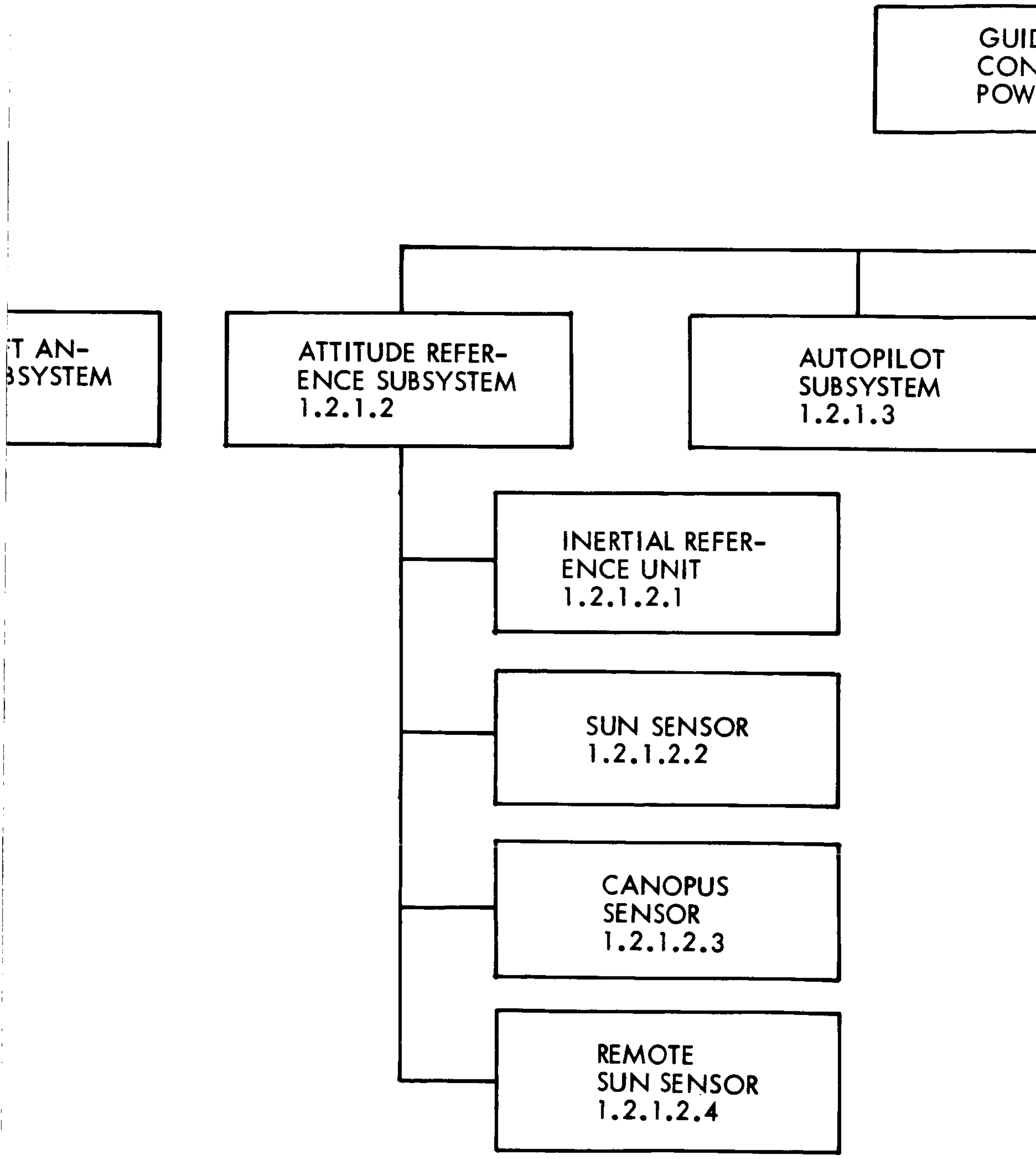
RELAY ANTENNA (VHF) 1.2.1.1.3.5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INERTIAL REFER-
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1.2.1.2.4

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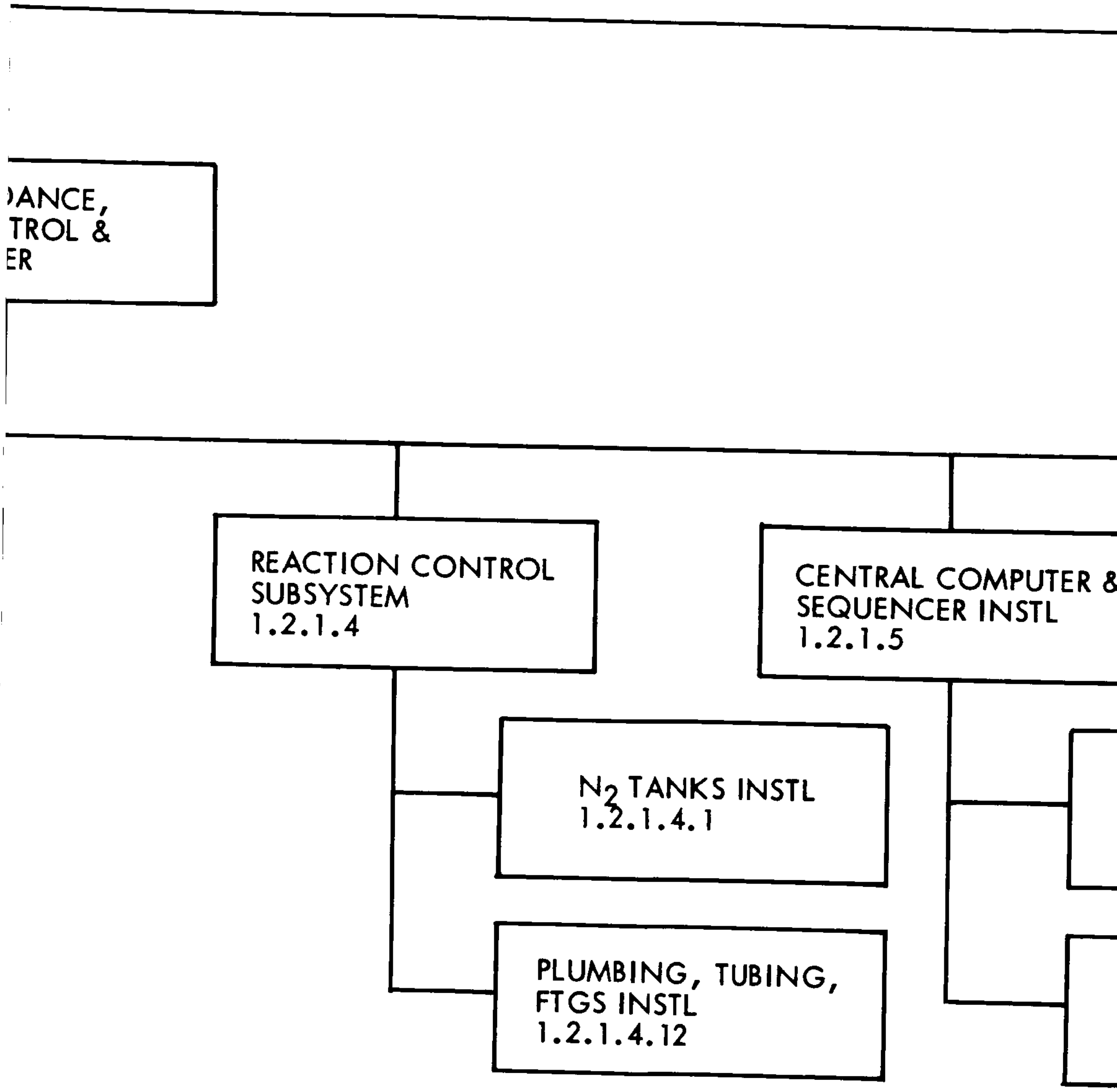
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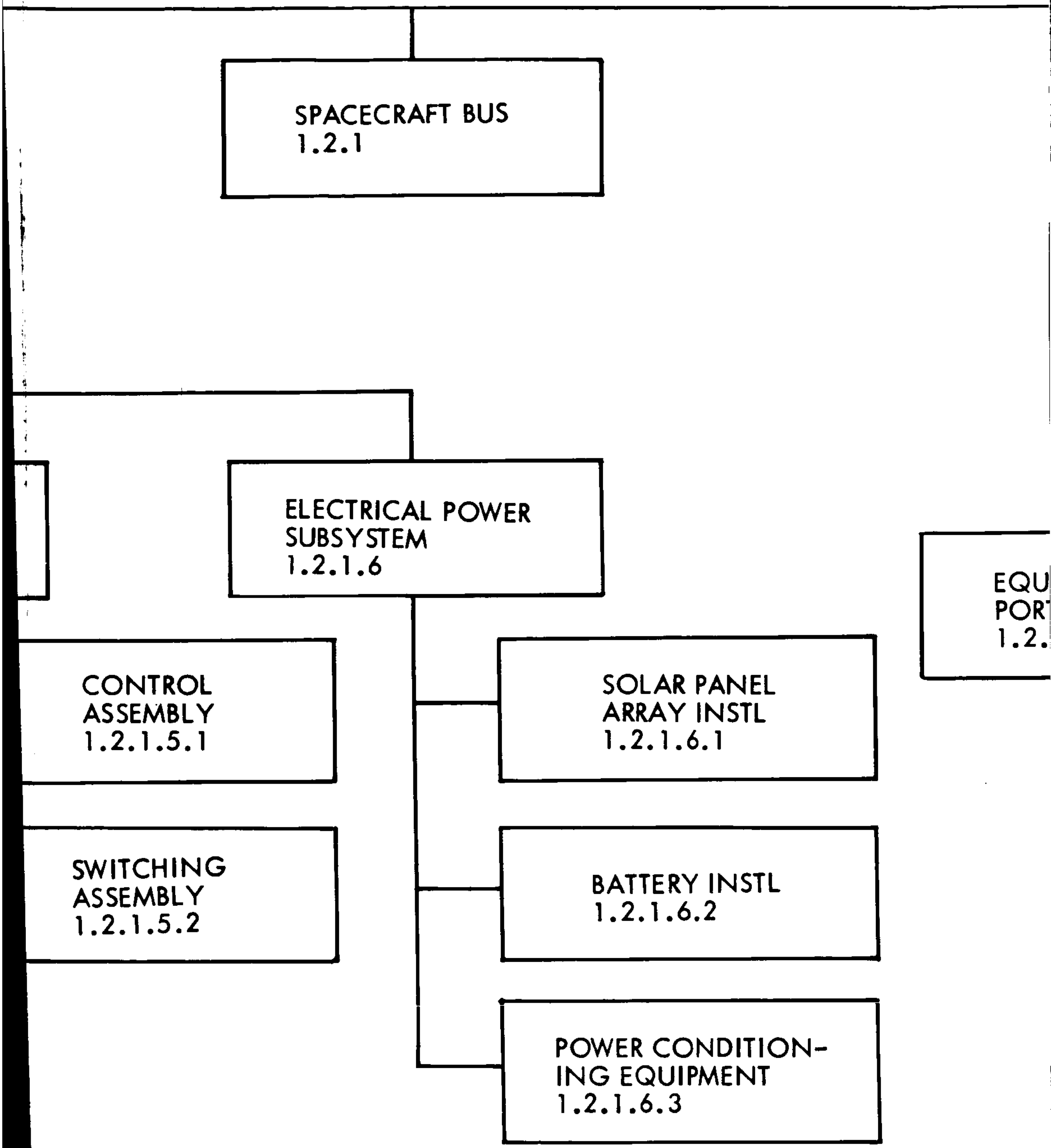
1.2.1.4.1
PLUMBING, TUBING, FTGS INSTL
1.2.1.4.12

98 3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SWITCHING ASSEMBLY 1.2.1.5.2
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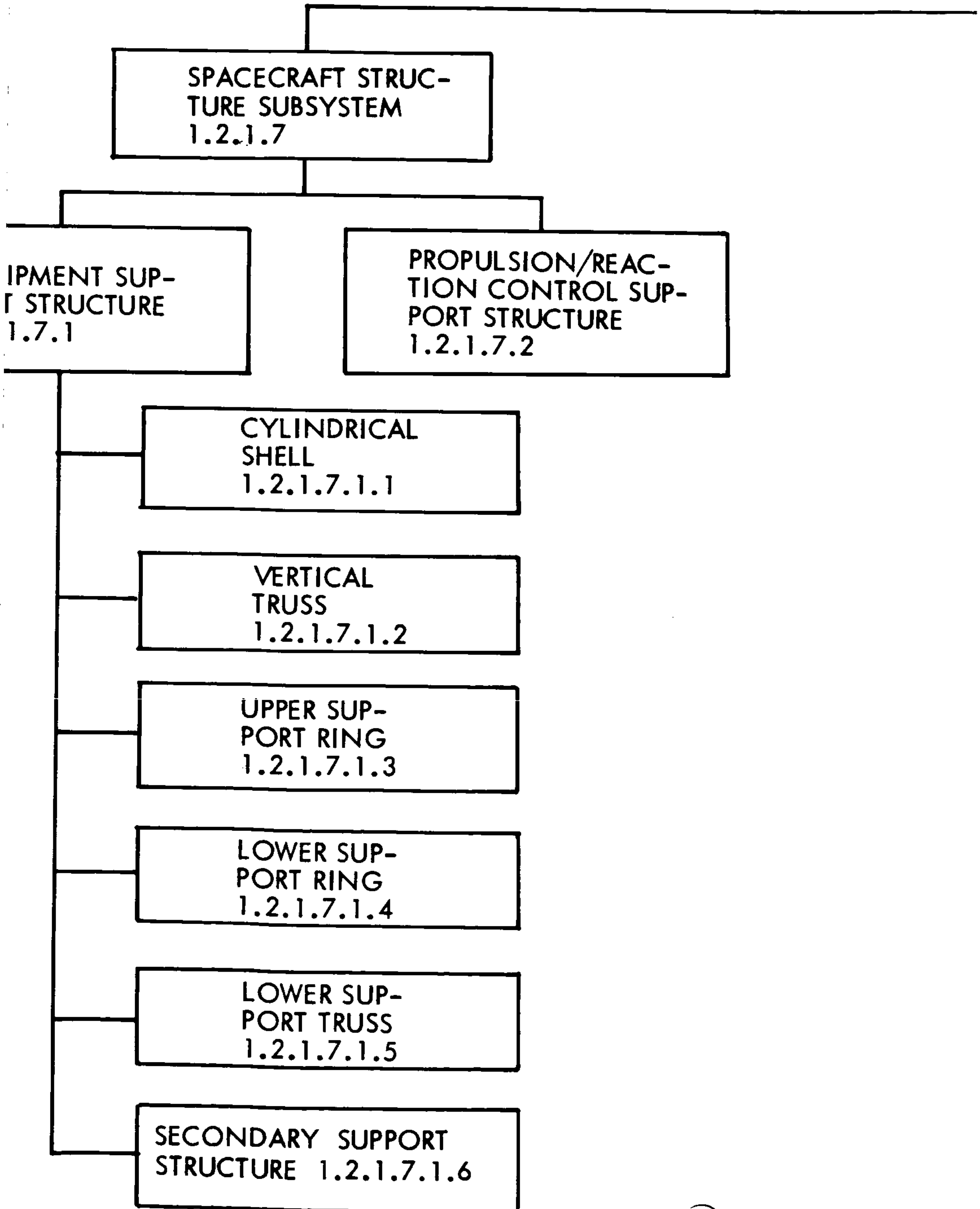
BATTERY INSTL 1.2.1.6.2
POWER CONDITIONING EQUIPMENT 1.2.1.6.3

98 7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1.7.1
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CYLINDRICAL SHELL 1.2.1.7.1.1
VERTICAL TRUSS 1.2.1.7.1.2
UPPER SUPPORT RING 1.2.1.7.1.3
LOWER SUPPORT RING 1.2.1.7.1.4
LOWER SUPPORT TRUSS 1.2.1.7.1.5
SECONDARY SUPPORT STRUCTURE 1.2.1.7.1.6

98 5

- HIGH-GAIN ANTENNA MECH INSTL 1.2.1.8.1
  - BOOST SUPPORTS 1.2.1.8.1.1
  - DEPLOYMENT LINKAGE 1.2.1.8.1.2
  - DEPLOY PIN PULLERS 1.2.1.8.1.3
  - DISH PIN PULLERS 1.2.1.8.1.4
  - POINTING MECHANISMS 1.2.1.8.1.5
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  - DE ME 1.2
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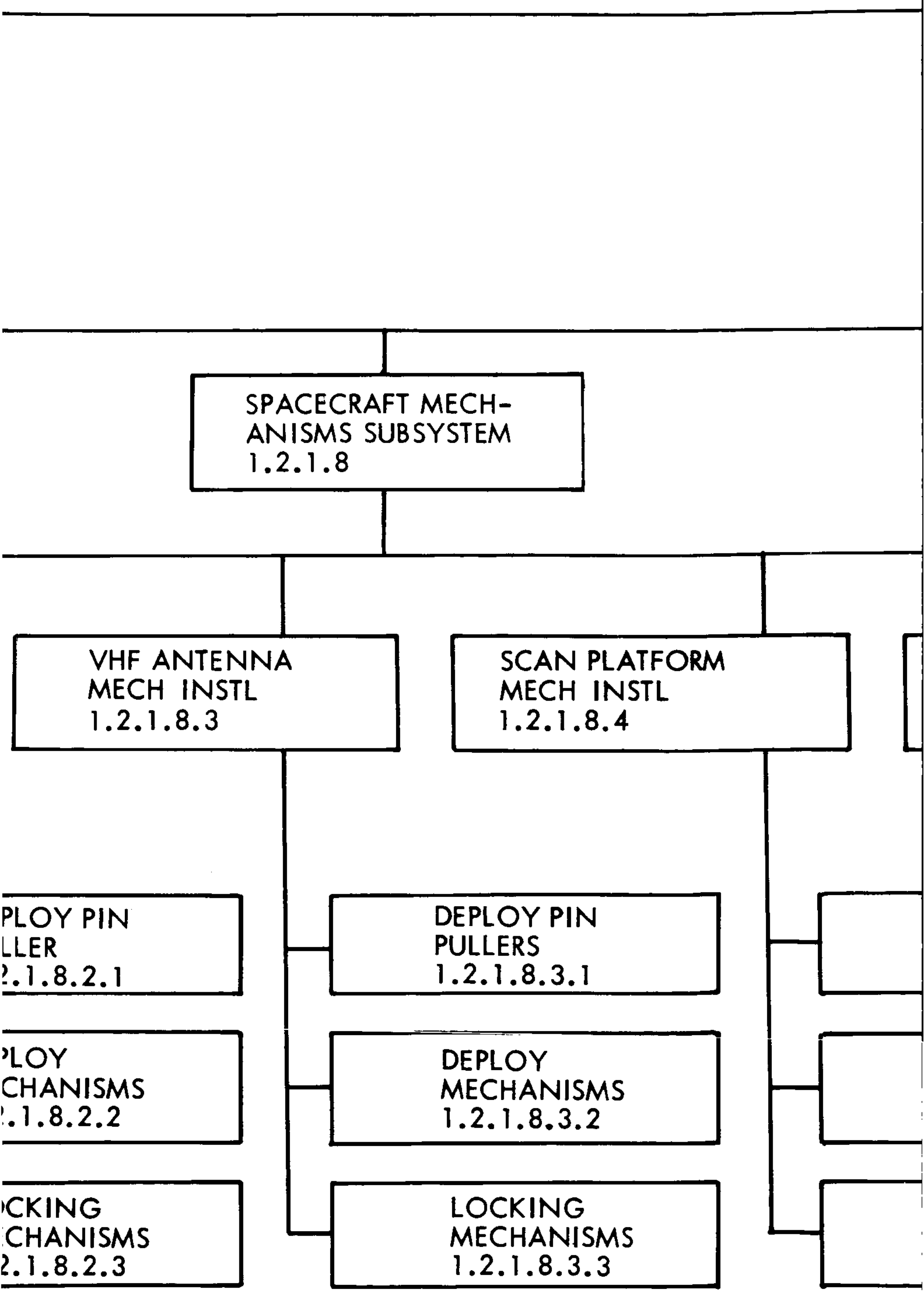
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MECH INSTL
1.2.1.8.4
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2.1.8.2.1
PLOY
CHANISMS
2.1.8.2.2
CKING
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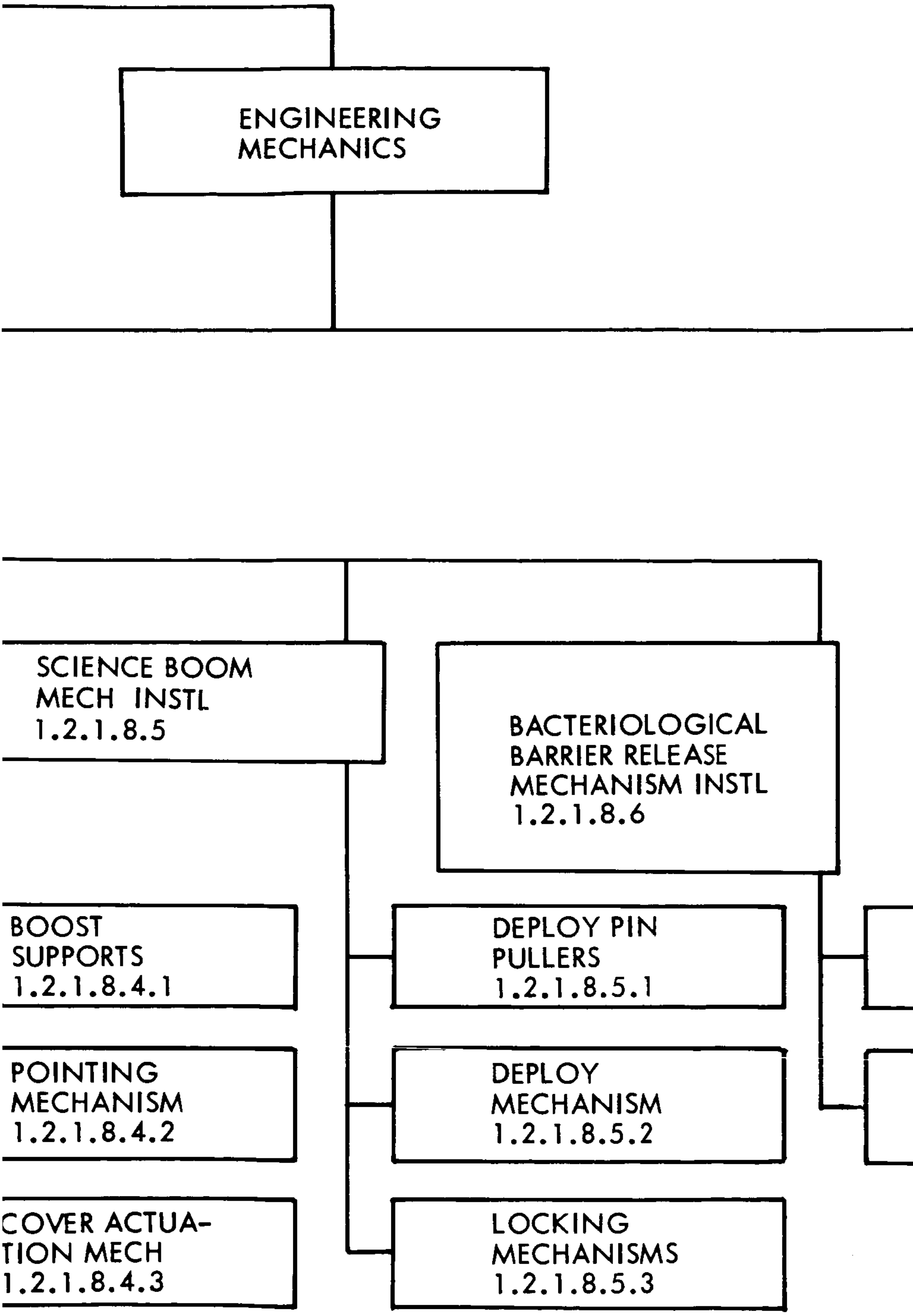
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POINTING MECHANISM 1.2.1.8.4.2
COVER ACTUATION MECH 1.2.1.8.4.3
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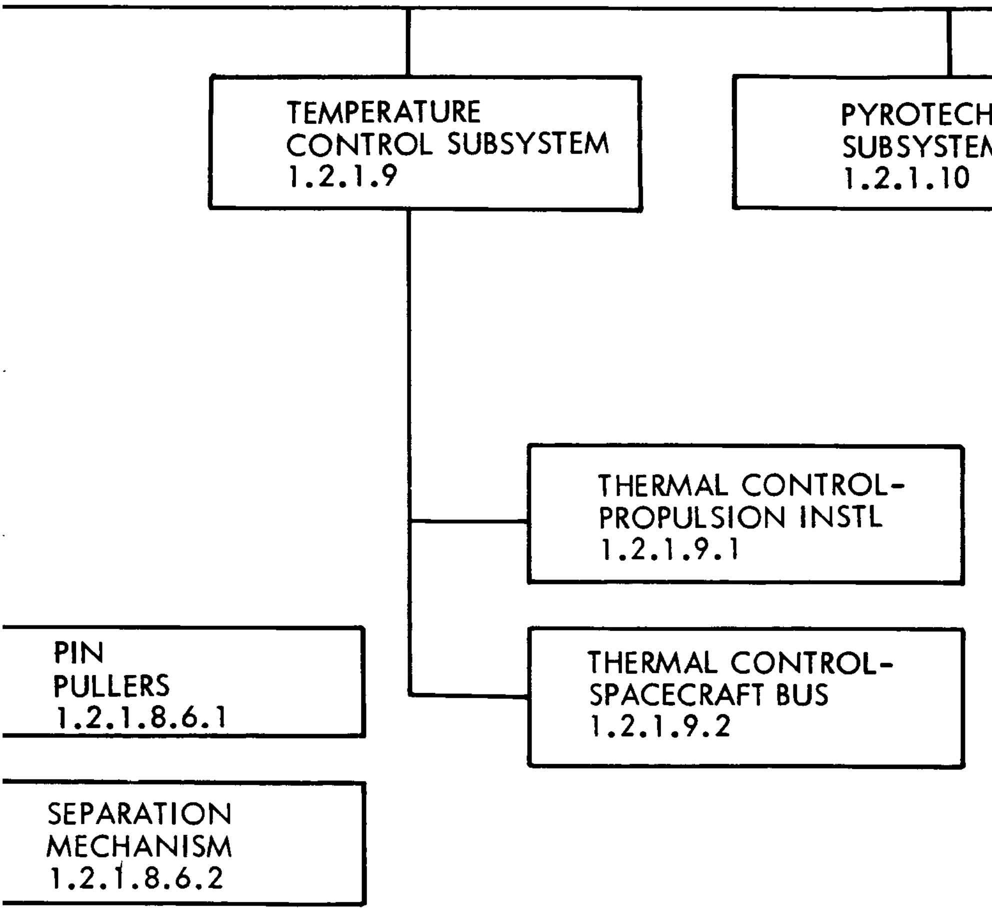
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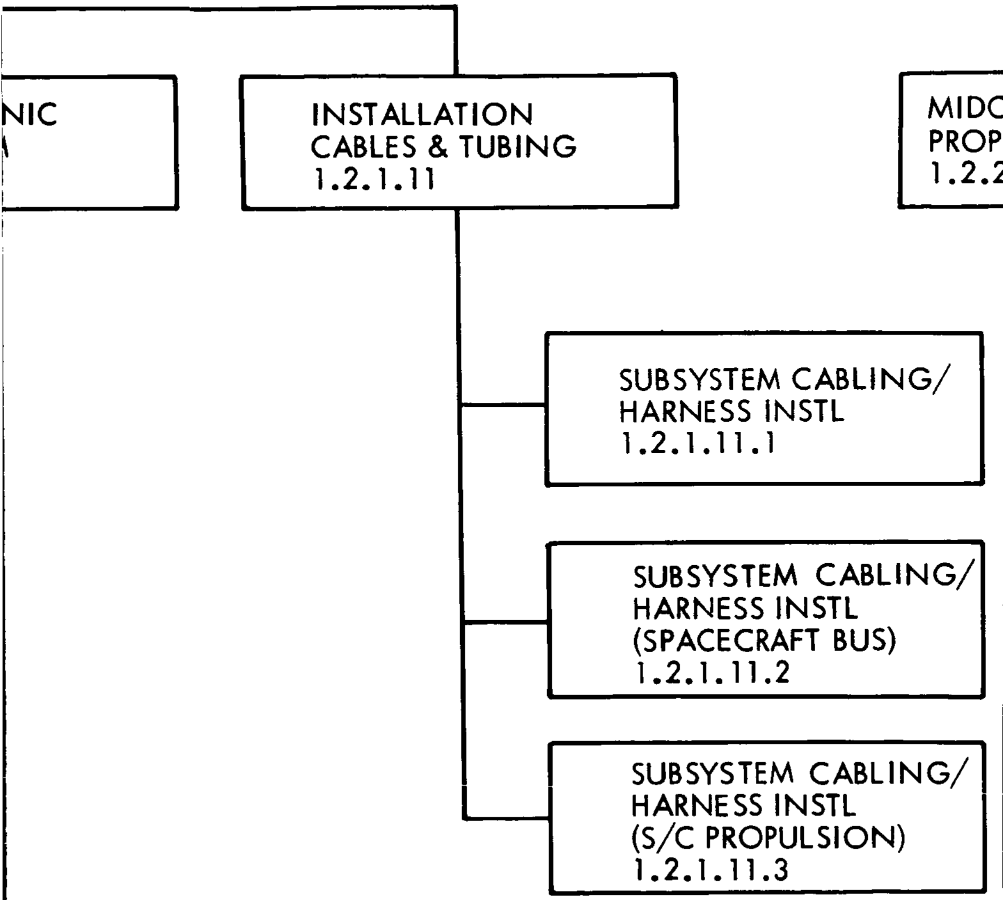
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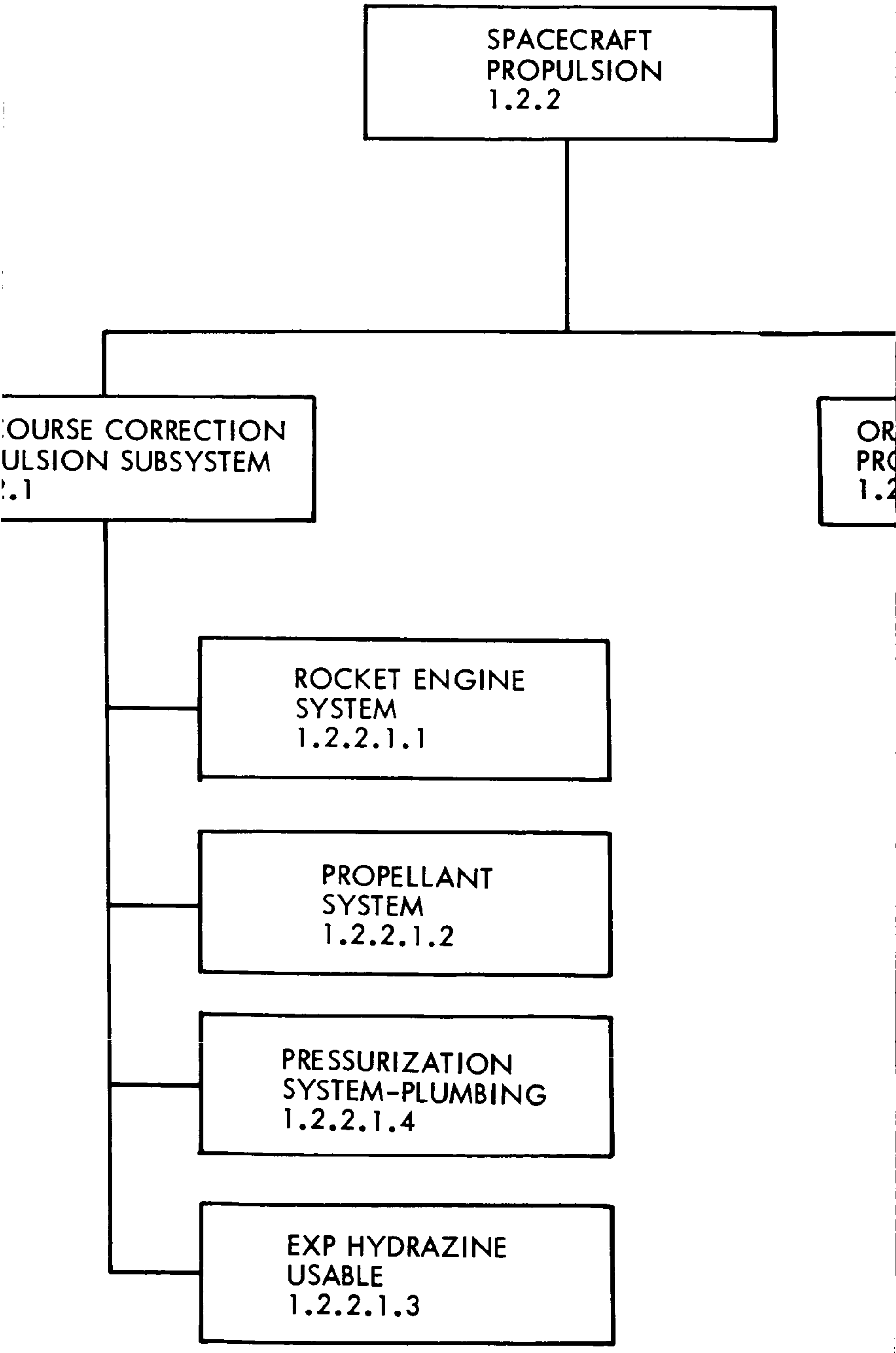
MIDC PROP 1.2.2
SUBSYSTEM CABLING/ HARNESS INSTL 1.2.1.11.1
SUBSYSTEM CABLING/ HARNESS INSTL (SPACECRAFT BUS) 1.2.1.11.2
SUBSYSTEM CABLING/ HARNESS INSTL (S/C PROPULSION) 1.2.1.11.3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ULSION SUBSYSTEM
.1
OR
PR
1.2
ROCKET ENGINE
SYSTEM
1.2.2.1.1
PROPELLANT
SYSTEM
1.2.2.1.2
PRESSURIZATION
SYSTEM-PLUMBING
1.2.2.1.4
EXP HYDRAZINE
USABLE
1.2.2.1.3

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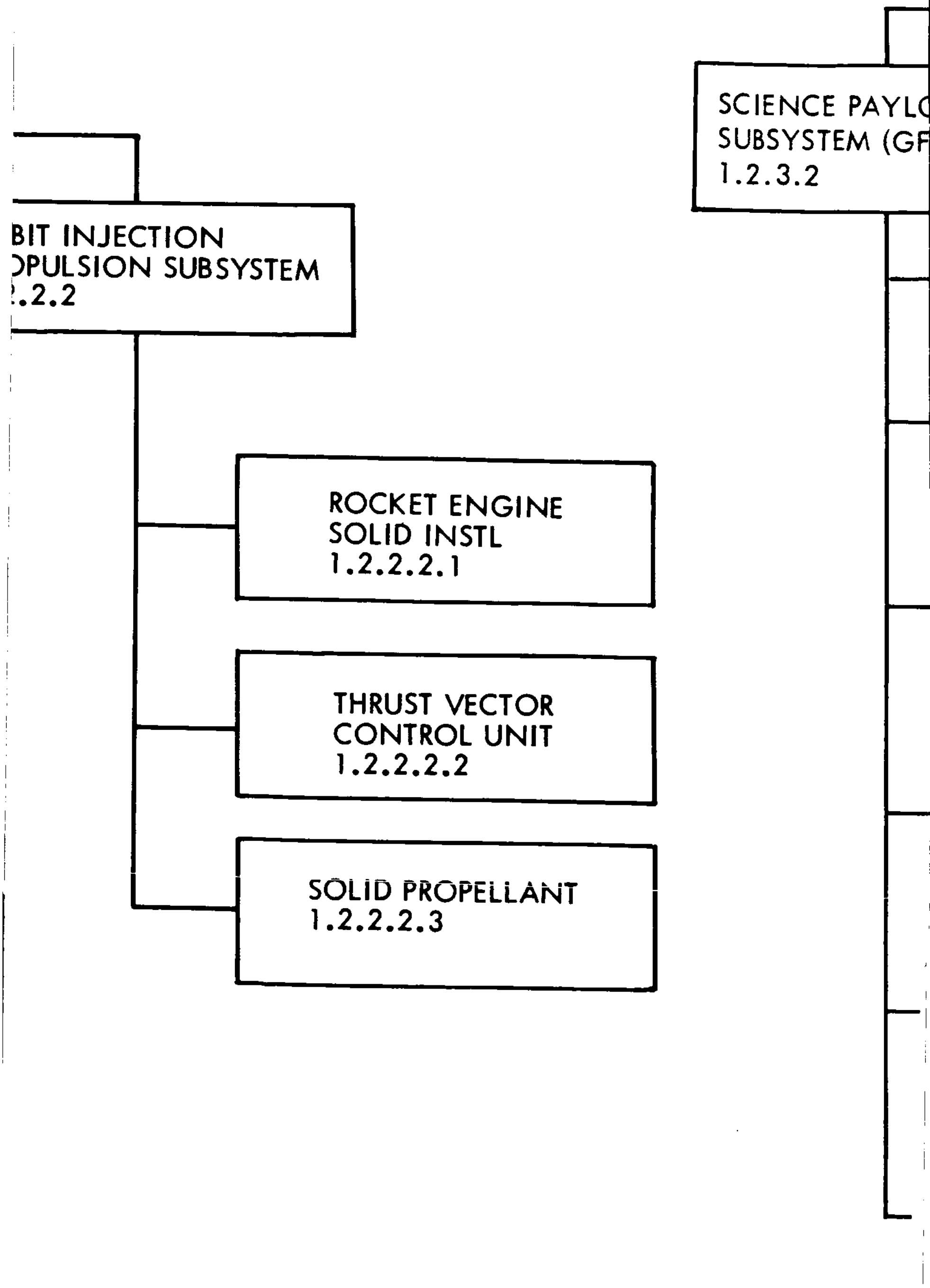
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PULSION SUBSYSTEM
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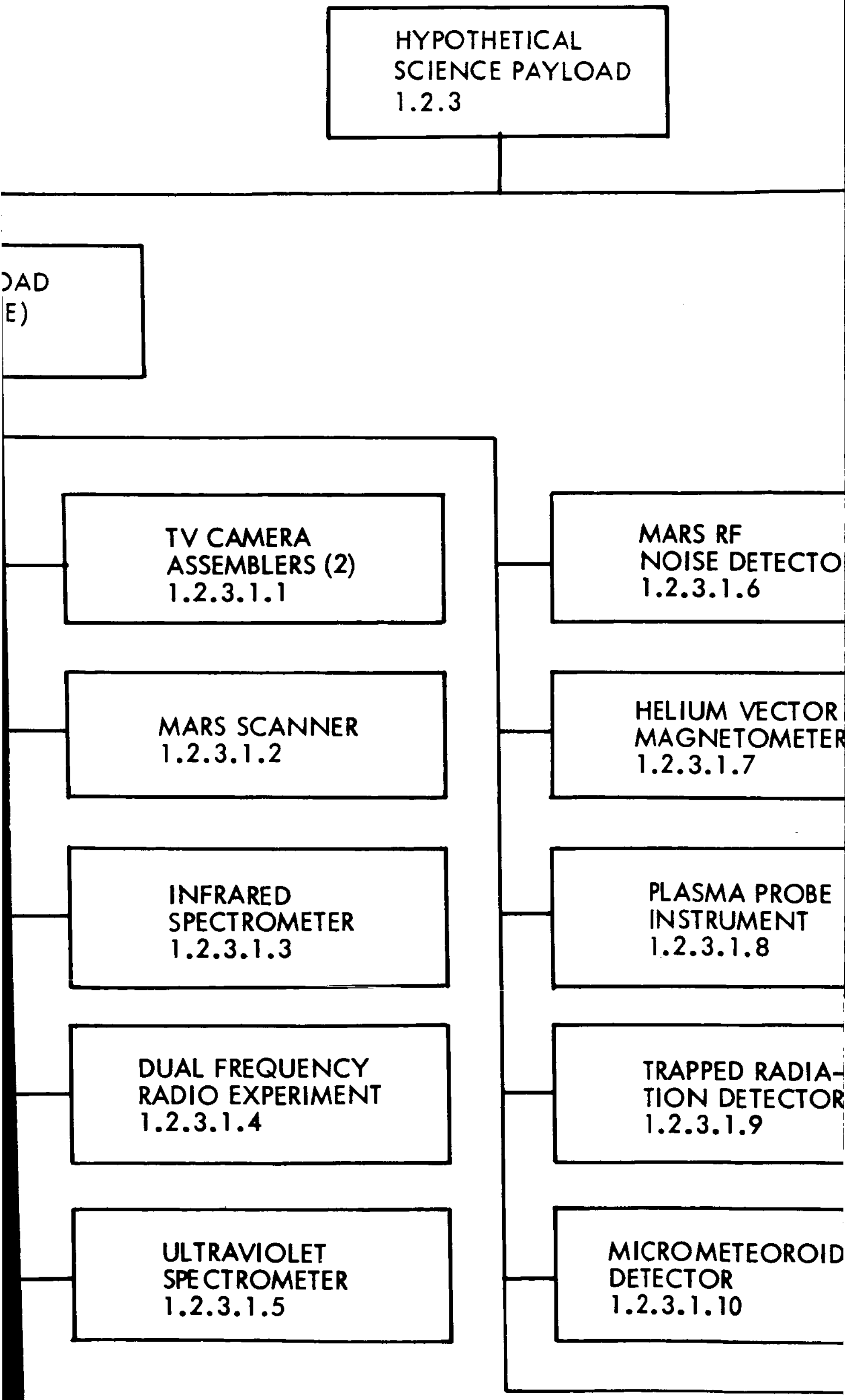
ROCKET ENGINE
SOLID INSTL
1.2.2.2.1
THRUST VECTOR
CONTROL UNIT
1.2.2.2.2
SOLID PROPELLANT
1.2.2.2.3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OAD
E)
TV CAMERA ASSEMBLERS (2) 1.2.3.1.1
MARS SCANNER 1.2.3.1.2
INFRARED SPECTROMETER 1.2.3.1.3
DUAL FREQUENCY RADIO EXPERIMENT 1.2.3.1.4
ULTRAVIOLET SPECTROMETER 1.2.3.1.5
MARS RF NOISE DETECTO 1.2.3.1.6
HELIUM VECTOR MAGNETOMETER 1.2.3.1.7
PLASMA PROBE INSTRUMENT 1.2.3.1.8
TRAPPED RADIATION DETECTOR 1.2.3.1.9
MICROMETEOROID DETECTOR 1.2.3.1.10

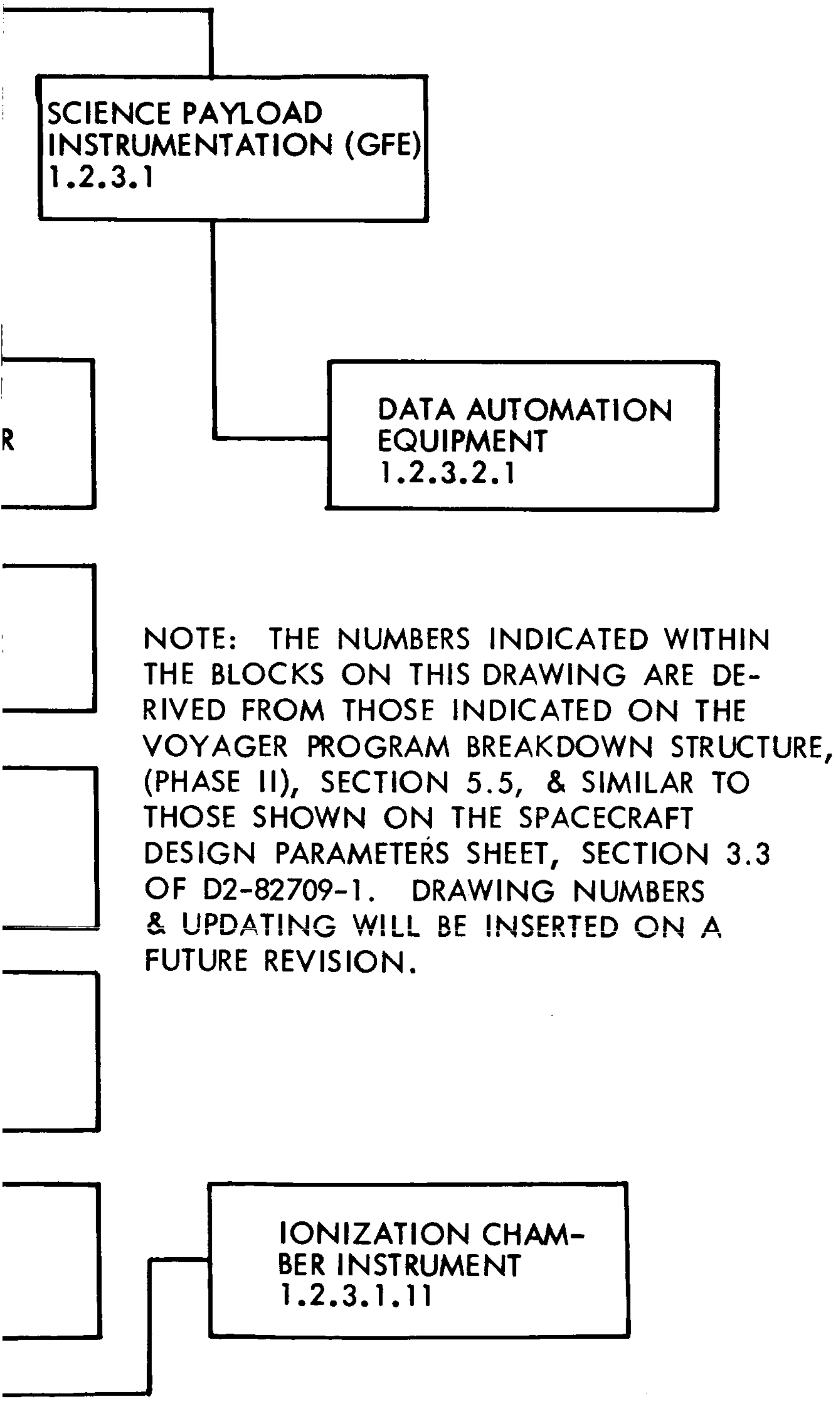


Figure 3.4-1: Equipment List/Drawing Tree

3.5 FLIGHT EQUIPMENT/LAUNCH VEHICLE INTERFACE REQUIREMENTS

3.5.1 Scope

This subsection describes the interface between the Planetary Vehicle and the launch vehicle. This interface definition includes consideration of the nose fairing, shroud, launch vehicle adapter, and their relationship to the Planetary Vehicle. It also describes Flight Spacecraft separation, dynamic interactions during flight, and launch vehicle performance.

3.5.2 Mechanical

3.5.2.1 General Description

The Saturn IB/Centaur launch vehicle configuration is shown in Figure 3.5-1. The interface between the Planetary Vehicle and the launch vehicle is a field joint at the interface between the spacecraft and the 120-inch launch vehicle adapter. (Saturn Station 2048). The interface consists of a mechanical joint and electrical connectors that accommodate any required operation between the Centaur and spacecraft stages.

3.5.2.2 Field Joints and Adapters

Field Joints--The field joints (Figure 3.5-2) permit the Planetary Vehicle to be handled in an encapsulated condition and will be designed to minimize the complexity of field operation.

Spacecraft-to-Flight-Capsule Adapter--The adapter (Figure 3.5-2) is located at Station 2107. The Flight Capsule is supported at the top of the Flight Spacecraft equipment support structure upper frame.

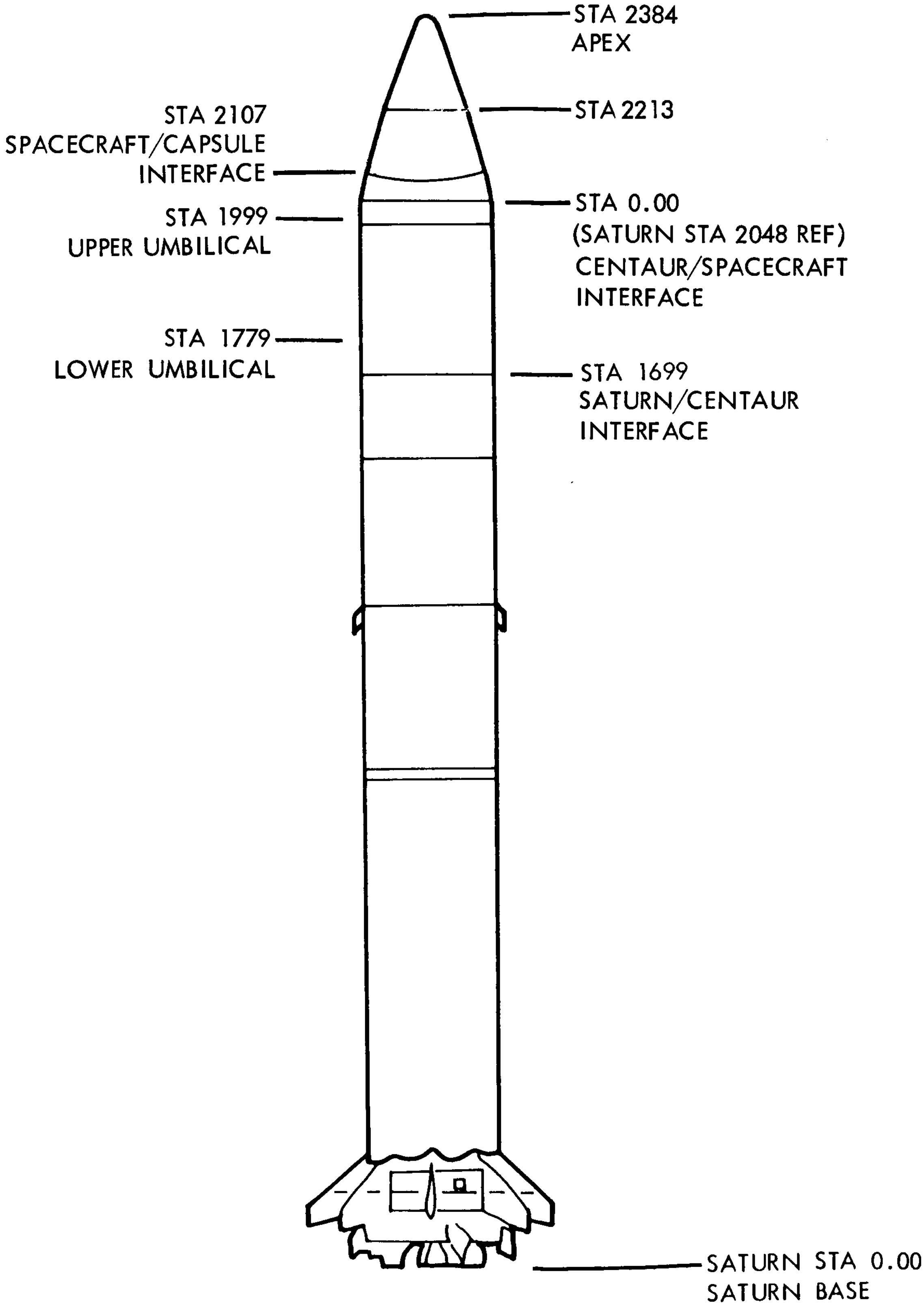


Figure 3.5-1: Space Vehicle

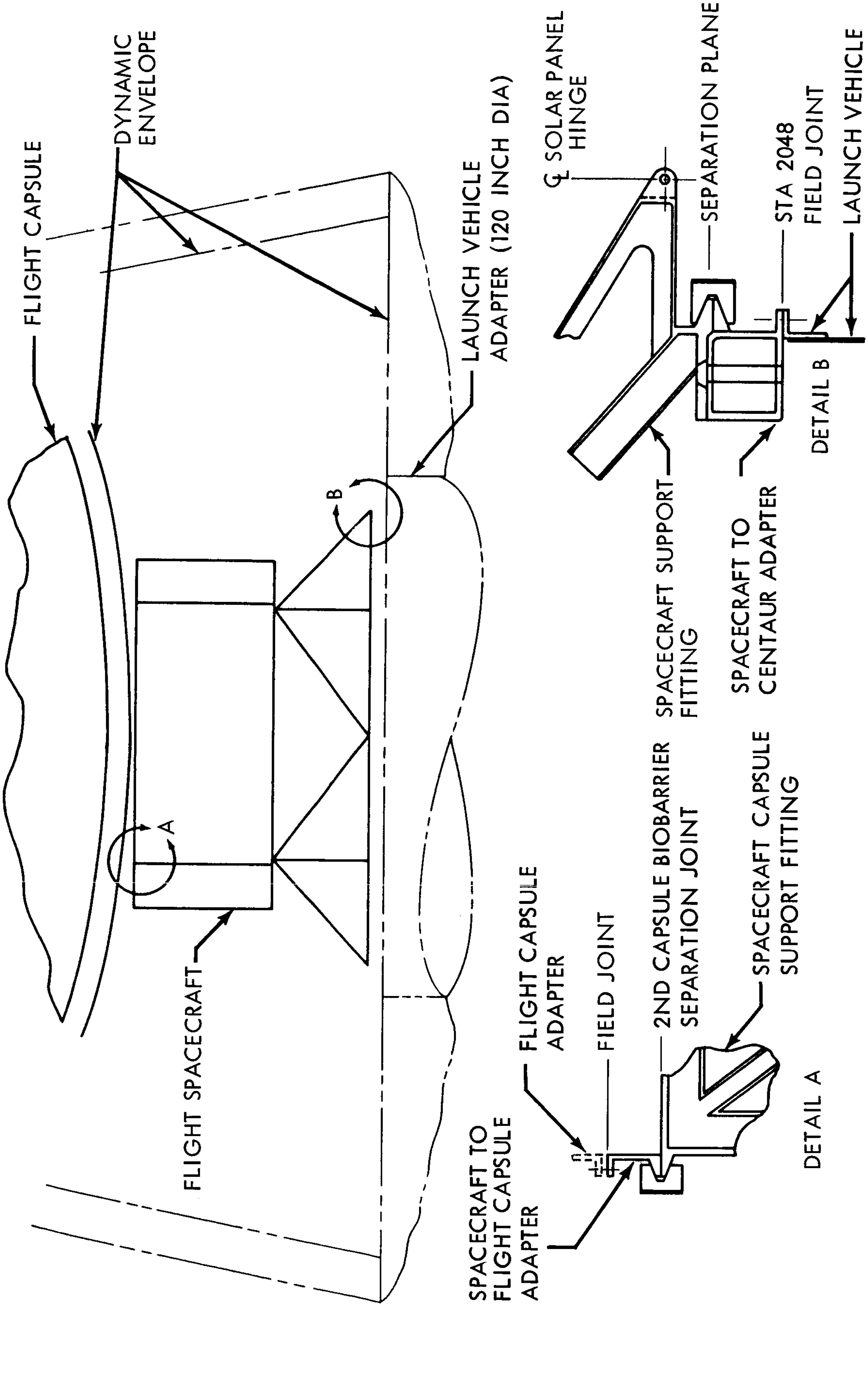


Figure 3.5-2: Field Joints & Adapters

Spacecraft-to-Centaur Adapter--A relatively short adapter from the spacecraft to the Centaur is required. The spacecraft also has a short ring at the base of the truss that mates to the adapter upper ring. The Flight Spacecraft is located above Station 2048. The Flight Spacecraft structure is octagonal in cross-section with eight vertical trusses at the apex and an upper and lower bulkhead. The Planetary Vehicle loads are distributed to the adapter by the eight vertical trusses.

Launch-Vehicle Adapter--The launch-vehicle adapter is located below Station 2048. The launch-vehicle adapter is the responsibility of the launch vehicle contractor. The adapter, a cylindrical structure mounted on the Centaur forward bulkhead ring, reacts to a combination of axial compression loads and bending moments developed by the payload. The adapter also acts as a support point through radial-mounted bulkhead-support cables between the adapter and the Centaur payload ring for maintaining tension on the Centaur forward bulkhead in the event that tank pressure is lost.

3.5.2.3 Shroud

Shroud Description--The shroud (Figure 3.5-3) is a 260-inch-diameter cylindrical structure mounted on the Saturn IB and extends to the upper level of the Centaur launch-vehicle adapter. The shroud has been designed to handle a combination of axial loads, bending moment, and crushing pressure. Snubbing and tensioning mechanisms are placed between the launch-vehicle adapter and shroud to restrain lateral motion of the Centaur and Planetary Vehicle, provide structural integration between shroud and tank, and apply a tensile force to the tank. Figure 3.5-4 shows the structural load paths from the Planetary Vehicle to the booster during

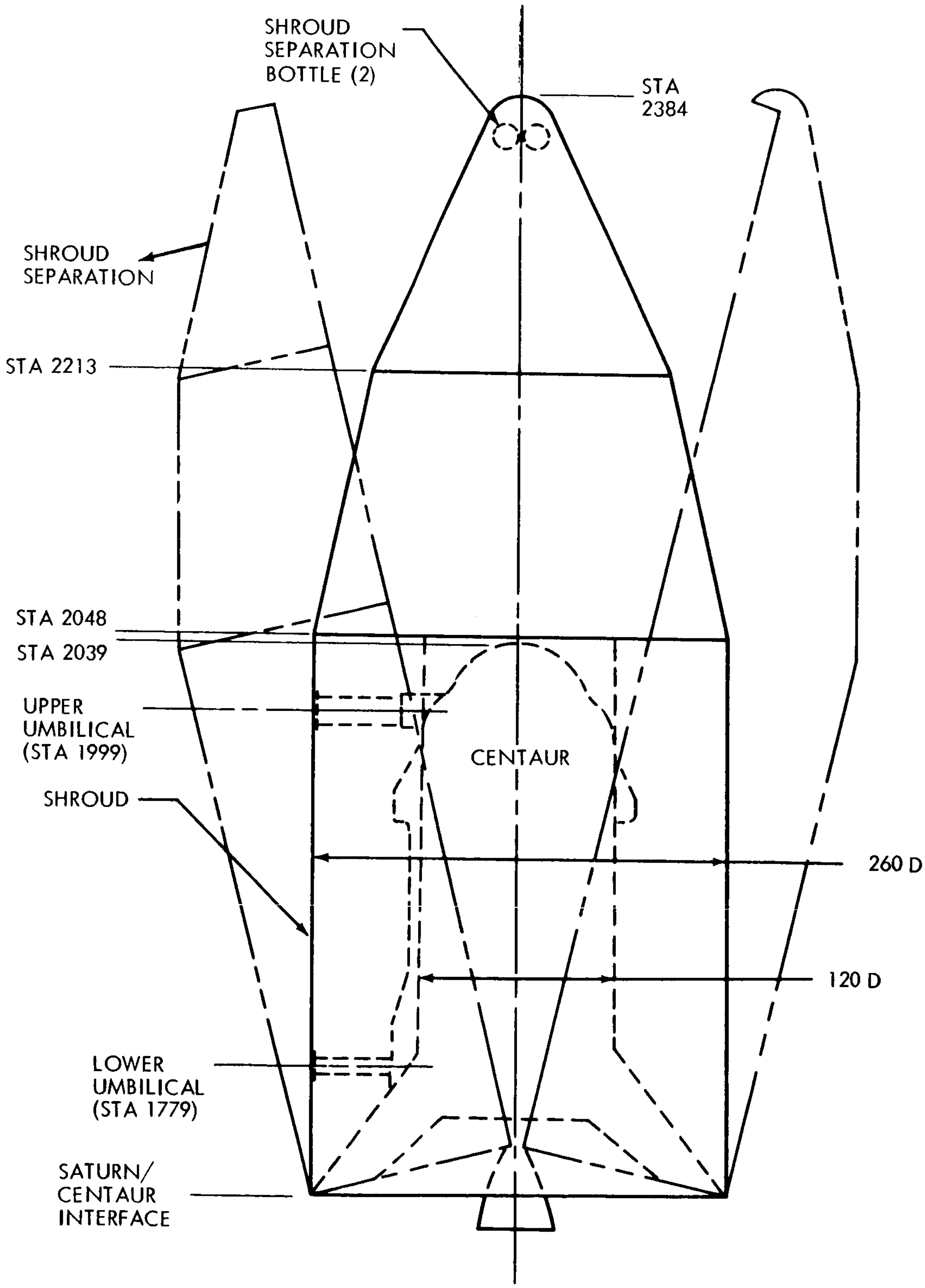


Figure 3.5-3: Centaur Shroud & Payload Fairing Assembly

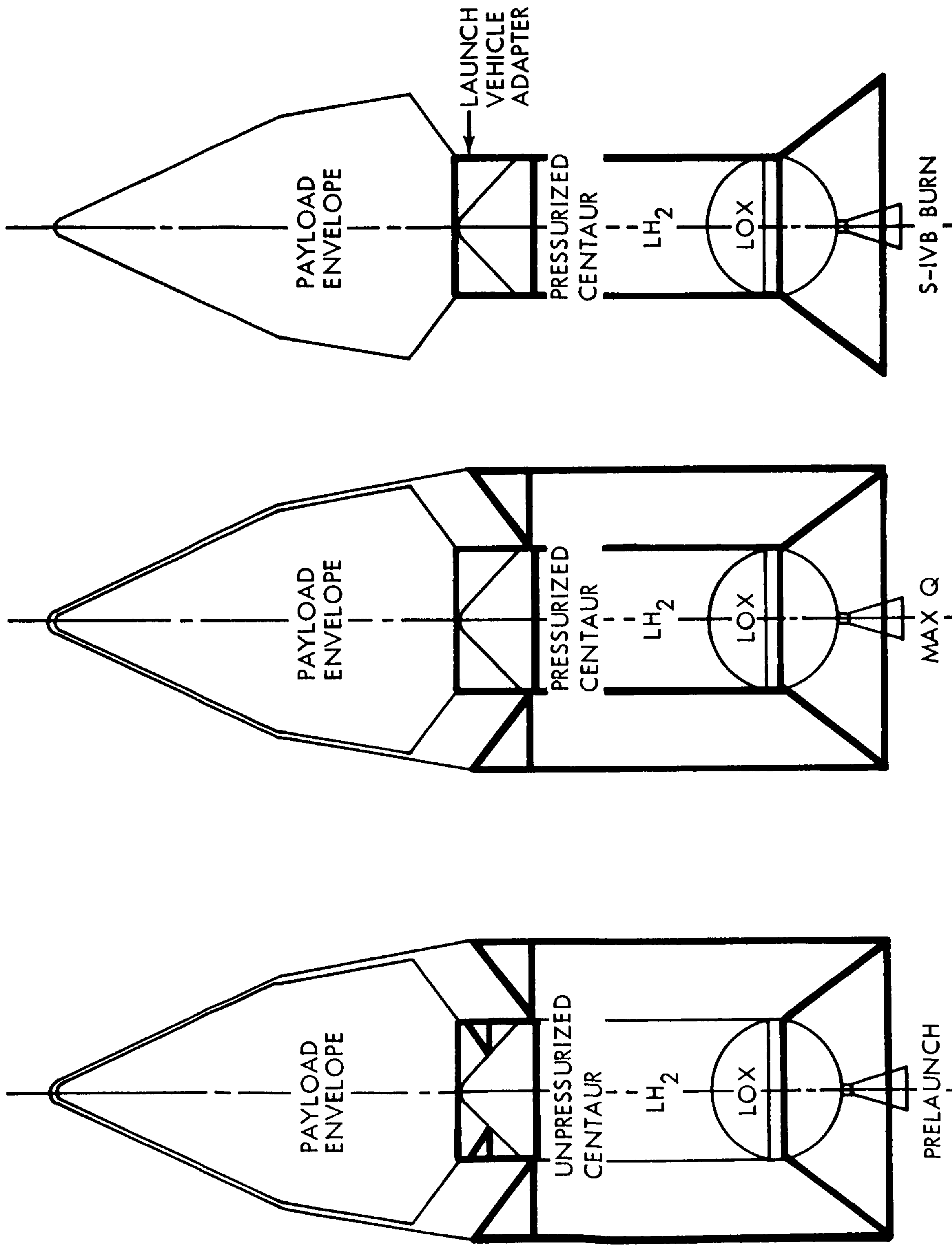


Figure 3.5-4: Centaur And Shroud Structural Load Paths

prelaunch and boost phases. If pressure is lost, the Centaur and Planetary Vehicle weight is transmitted through the launch-vehicle adapter and tensioning mechanisms to the shroud. Springs on the snubbing rods compensate for tank contractions or expansions during pressure changes or cryogenic loading. Two umbilical tunnels are located between the shroud and Centaur as shown in Figure 3.5-3. The payload fairing consists of a nose cap and two conical frustrums that are attached to the forward end of the 260-inch shroud (as described in "Description and Status of Saturn IB," presented to American Astronautical Society at Denver, Colorado, February 8-10, 1965, by F. L. Williams).

Shroud Ventilation--A gaseous-nitrogen shroud purge maintains an inert atmosphere in the plenum area between the 260-inch diameter nose fairing and the Centaur and in the area around the electronic components located on the forward liquid-hydrogen bulkhead. The purge is introduced at two points at the start of propellant loading and continues until lift-off. It enters in the area of the electronic components (1000 scfm at 40°F). A physical barrier is placed around the electronic components to ensure that no gases will enter this area. The first purge is introduced into the physical barrier and flows from there into the shroud plenum area. The barrier contains orifices that maintain a slight positive pressure in the electronic components area. The shroud-vent area required during ground purge is sufficient for venting during the ascent portion of the flight.

Shroud Separation--The fairing and shroud are longitudinally separated (Figure 3.5-3) into clamshell halves by linear-shaped charges in conjunction with ringframe segmentation by explosive bolts. Explosive bolts also separate the snubbing and tensioning mechanisms from the Centaur tank. Simultaneously, linear-shaped charges sever the shroud circumferentially, just forward of the aft ring frame. Cold nitrogen-gas jets from two pressurized spheres, located in the most forward portion of the payload fairing, force the segments to pivot outward about the base hinge points. The cold-flow thrust, in conjunction with increasing angular velocity (as a consequence of launch vehicle acceleration due to SIB thrust), frees the segments from their base pivot points.

Environmental Control--Conditioned air is needed for the Voyager spacecraft during prelaunch (up to the time of lift-off). The launch vehicle contractor (Centaur) has responsibility to provide the required environmental control after Planetary Vehicle Launch Vehicle mating. Coordination between the spacecraft and the launch vehicle contractor is necessary to ensure meeting the thermal requirements.

3.5.2.4 Clearances

The basic shroud clearance requirement is that the shroud not encroach on the spacecraft envelope (Figure 3.5-5) either during powered flight or shroud separation. No requirement is specified for any additional clearance in shroud dispersion, absolute dimensions, or factor of safety. Collision skids are not required on the spacecraft since a clamshell nose fairing is used rather than an over-nose fairing. The motion of the fairing does not present undue collision hazards.

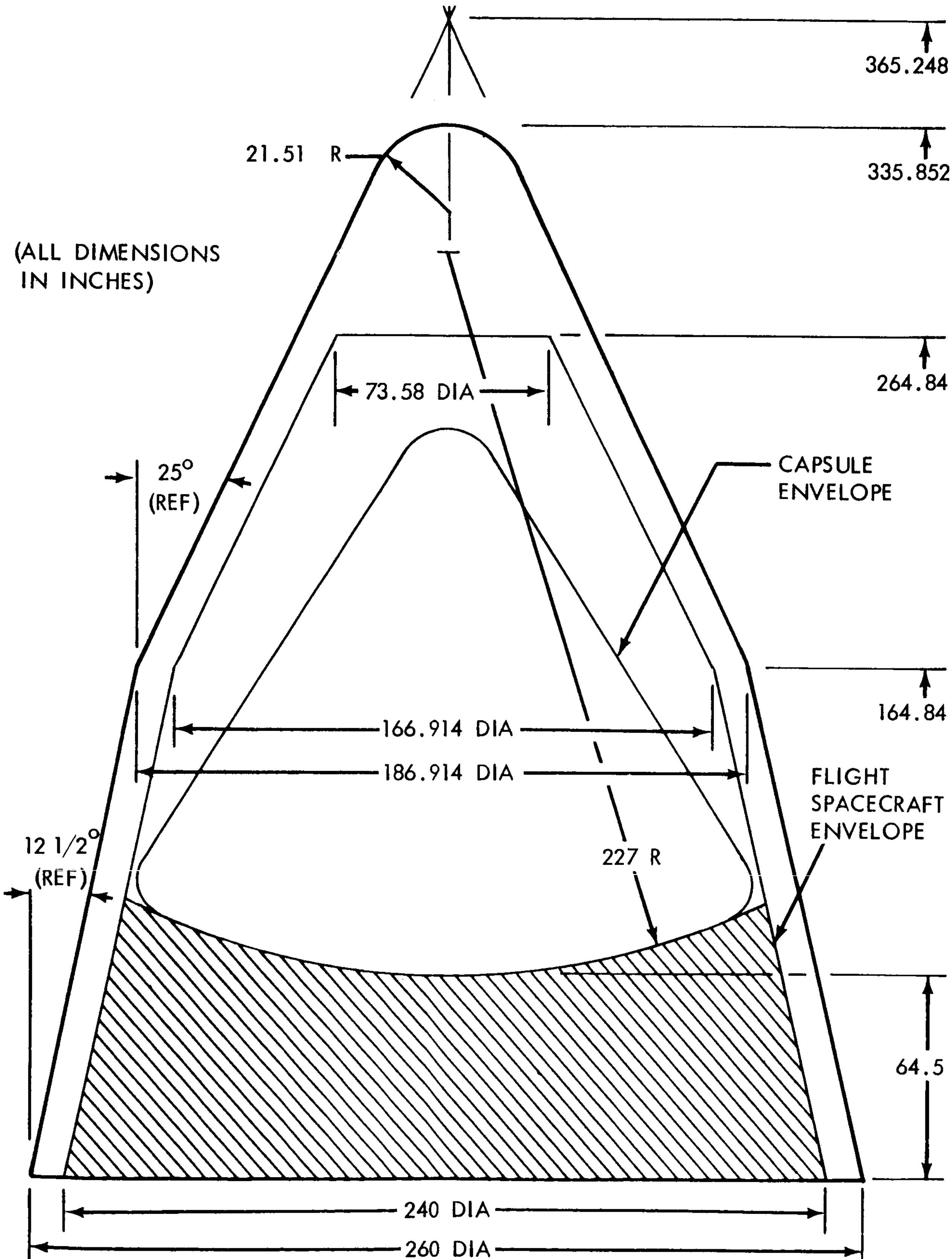


Figure 3.5-5: Spacecraft Envelope

3.5.3 Electrical

3.5.3.1 General Description

The spacecraft and Centaur electrical interface is shown schematically in Figure 3.5-6. The electrical interface includes:

1) RF coupler, ascent antenna and parasitic antenna in the Centaur shroud;
2) Electrical disconnects;
3) Spacecraft to umbilical electrical connections.

3.5.3.2 Umbilical Connections

There are three umbilical connections on the Centaur booster at Station 1779, 1999, and 2048. The upper umbilical (Station 1999) accommodates all Planetary Vehicle hardlines connections to the launch complex through a catenary cable assembly. The lowest umbilical (Station 1779) is used for fueling the Centaur booster. The third umbilical (Station 2048) is required for the interface functions between the Centaur and the spacecraft.

3.5.3.3 Telemetry

There are two telemetry interfaces between the spacecraft and Centaur.

1) Electrical signal connection through the (Saturn Station 2048) connector which relays signals to the Centaur VHF transmission system. (See Section 3.5.6.2)
2) Spacecraft S-band transmission via the ascent antenna located below Saturn Station 2048.

The latter interface consists of a Boeing-furnished S-band antenna and a matching parasitic antenna located in the Centaur shroud. The coaxial cabling for this antenna includes an RF coupler at the interface (Saturn Station 2048). The ascent antenna separates from the Planetary Vehicle at the RF coupler when Centaur separation occurs.

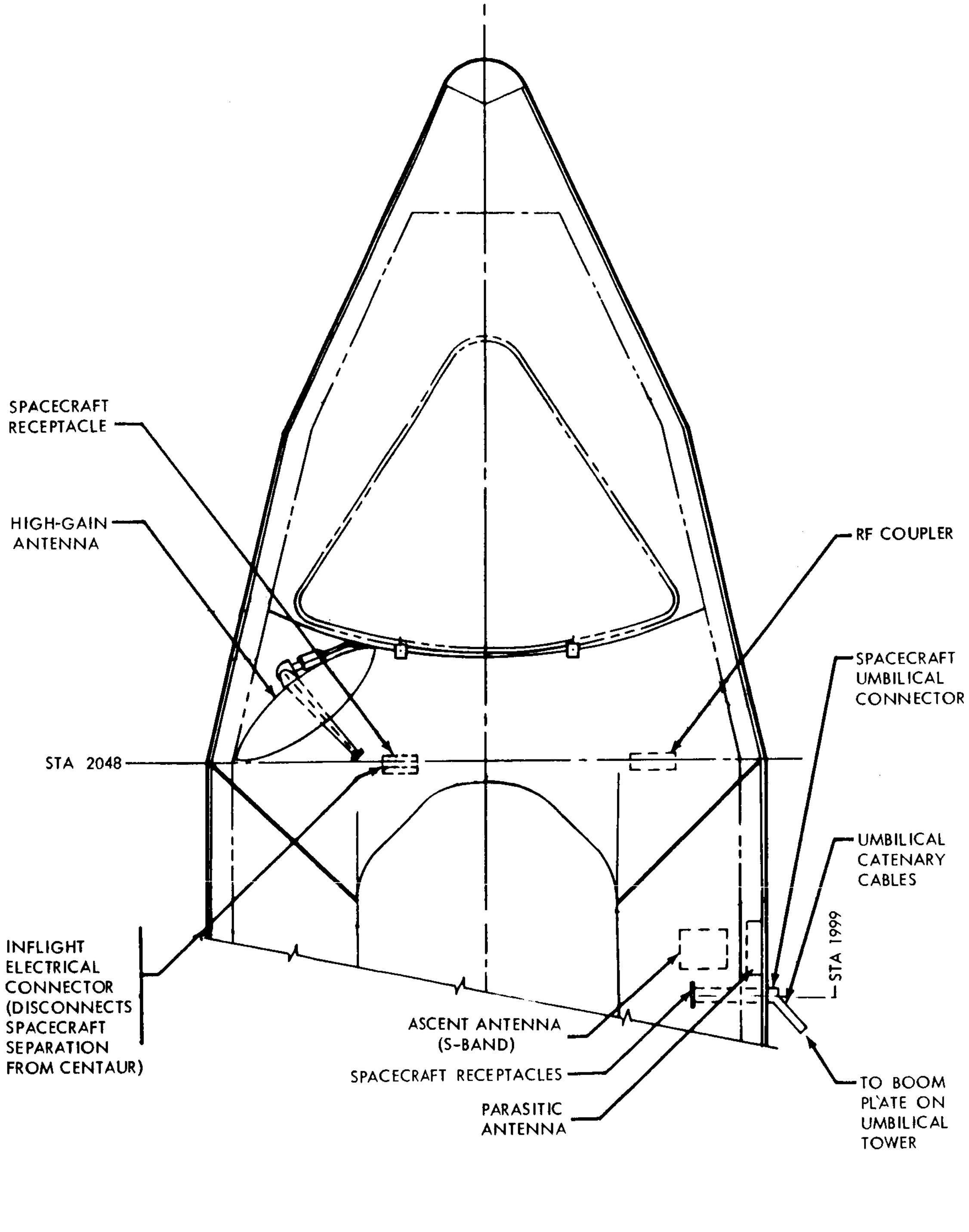


Figure 3.5-6: Electrical Interface

3.5.4 Centaur/Spacecraft Separation

3.5.4.1 Description of Separation

Separation of the Planetary Vehicle from the Centaur is initiated by a signal from the Centaur to the spacecraft pyrotechnic subsystem. The interfaces to be separated include the mechanical interface at the junction of the launch vehicle and spacecraft adapter, and the electrical connectors located at the spacecraft-to-adapter interface.

3.5.4.2 Dynamic Limitations

The dynamic limitations during and after separation of the spacecraft require that the following criteria be satisfied:

1) The period between Centaur-burn termination and spacecraft separation must be a minimum of 10 minutes to ensure that Centaur propulsion transients have subsided and that the spacecraft is separated from an attitude-stabilized vehicle;
2) The angular velocity imparted to the Planetary Vehicle, including Centaur residual angular rate, must not exceed 3 degrees per second about any Flight Spacecraft axis and the angular rate shall not exceed 1 degree per second has been a design goal.
3) Following separation, the differential velocity between the Planetary Vehicle and the Centaur will be a minimum of 2 feet per second (0.61 meter per second). This requirement is intended to preclude acquisition of the Centaur by the spacecraft Canopus sensor;
4) The separation forces applied to the Planetary Vehicle must be limited to 47,000 pounds (21.3×10^3 kg) to avoid the possibility of spacecraft damage;

5) Incremental velocity magnitude uncertainty imparted to the Planetary Vehicle must not exceed ± 15 feet per second (± 4.78 meters per second).

3.5.5 Launch Vehicle Performance Requirements

3.5.5.1 Injection Energy

The Saturn S-IB is required to deliver an 8300-pound-minimum (3760 Kg) payload to the injection altitude with a C_3 of at least 18 km^2/sec^2. A 100-nautical-mile parking orbit is used. Its inclination is variable to obtain the desired daily launch window within the launch azimuth constraints imposed by tracking and range safety considerations. Detailed injection requirements are contained in Section 2.0 of this document.

3.5.5.2 Injection Accuracy

Smaller spacecraft midcourse corrections can be obtained by accurate injection guidance. The Launch Vehicle shall not require the Flight Spacecraft to exceed a correction velocity of 75 meters per second. Error analyses conducted by the launch vehicle contractor will provide estimates of the root-mean square midcourse correction. Spacecraft (Centaur stage) aiming point at injection is biased to comply with the planetary quarantine constraint. The selected aiming point results in a Mars impact probability of 8×10^{-6}. The Mars impact probability allotted to the spent Centaur stage, however, is 5×10^{-6}. The Centaur stage is provided with retro-rockets. The post-injection retro-maneuver applied by the Centaur reduces its impact probability to less than the allotted value of 5×10^{-6}.

3.5.6 Inflight Instrumentation

3.5.6.1 Instrumentation

The extent of spacecraft-oriented instrumentation to be carried depends on Centaur telemetry system weight and launch vehicle performance margins. These margins, in turn, depend on the telemetry system (or modification thereof) that is selected. The minimum instrumentation is

1) Spacecraft separation measurements;
2) Environmental measurements.

3.5.6.2 Centaur Telemetry

The Centaur telemetry system continuously transmits the spacecraft composite modulation from time of launch until spacecraft separation. The data encoder and Centaur telemetry interface is a two-wire interface (one wire is the Centaur telemetry ground). The data-encoder output is transformer-isolated and varies between zero and five volts. Mixer output to the Centaur and spacecraft interface is a 400 to 100,000 bits-per-second pulse train.

3.5.6.3 Environmental Measurements

Monitoring of the acoustic levels within the shroud is desired during launch. Since several events of acoustic significance occur, a continuous telemetry channel is desired. The preferred location of the microphone is near the centerline of the shroud and above the solar panels.

Knowledge of both the low- and high- frequency vibration within the spacecraft is required. Three low-frequency and one high-frequency vibration transducers are positioned in the spacecraft at locations considered to be representative for structural load confirmation and equipment vibration levels.

3.5.7 Dynamic Interactions During Powered Flight

3.5.7.1 Dynamic Behavior

The dynamic motions of the shroud and spacecraft during powered flight will be analyzed. The objective of the analysis is to confirm that no contact between the shroud and the spacecraft occurs during powered flight. The dynamic reponse of the entire launch vehicle to the various launch events will be considered in the analysis. The dispersion of the shroud and spacecraft will be defined relative to a common point on the base of the adapter. The definition will include the effects resulting from longitudinal, lateral, and rotational motion at the base of the adapter caused by controlled-power flight of an elastic vehicle through the atmosphere superimposed on the effects caused by vibration. Fabrication and assembly tolerances will be included in the dispersions. Spacecraft characteristics used in analysis, such as elastic properties, mode shapes, and damping, will be calculated or derived from test data and will include thermal effects as appropriate. The required shroud, adapter, and Centaur equipment-rack properties will be provided by the launch vehicle contractor.

3.5.7.2 Shroud Separation Analysis

A dynamic analysis of shroud separation will be performed to determine the motion of the shroud during and after separation. The analysis will include, but not be limited to, the effects caused by:

1) Time variants in the action of explosive bolts, etc.;
2) Spring performance;
3) Spring geometries;
4) Center-of-gravity offset;
5) Residual rates;
6) Friction effects.

The conditions requiring analysis include:

1) Expected shroud separation based on the arithmetic sum of worst tolerances in spring forces, spring alignment, center-of-gravity offset, and residual rates;

2) Expected shroud dispersions based on the square root of the sum of the squares. The statistical character of the input distrubances is thus recognized and a statistical approximation of their effects is obtained.

3) Rational failure conditions;

4) Parametric investigation showing the effects of the individual input disturbances.

3.6 VOYAGER FLIGHT EQUIPMENT--TELEMETRY CRITERIA

3.6.1 Scope

The basic telemetry criteria to be applied in the design, construction, testing, and use of the telemetry and data storage subsystem are specified. The primary objective is that the data user have full confidence in the displayed data. The data must represent the actual performance of the spacecraft and the results of the all important scientific measurements made of space and planetary phenomena. This task is to be done as simply as possible. Criteria to be followed are found in the Mission Guidelines and Specifications and result from preliminary design studies. Factors considered are mission phase, measurement requirements (Section 3.10), reliability, power, weight, volume, environmental, and radio communication constraints.

3.6.2 Applicable Documentation

Military Specification MIL-I-6181D, "Interference Control Requirements, Aircraft Equipment"

Boeing Document D2-23834-1, Rev. A, "Voyager '71 Program Reliability Analysis and Prediction Standards"

3.6.3 Description

3.6.3.1 General

The Flight Spacecraft will be equipped with communication equipment capable of transmitting both Spacecraft Bus and capsule data to the DSIF. The transmitted data consists of capsule engineering and science data, bus engineering, cruise science, and planetary science data. After landing, capsule science data is transmitted to Earth and to the Spacecraft Bus. The bus telemetry and data storage subsystem (TDSS) retransmits the capsule data as a backup link to the capsule-to-Earth telemetry link. During critical maneuvers, capsule and bus engineering data are stored for retransmission at a higher data rate. After planet encounter, high-rate planetary science data is available simultaneously for both recording on tape for retransmission at a lower data rate and for direct real-time transmission.

The Voyager TDSS will use seven data subcarrier frequencies, eight data rates, and six transmission modes to accommodate the data transmission requirements associated with the various mission phases. Only two subcarriers are used at a time. An emergency mode is included for engineering data only.

3.6.3.2 Data Formatting

The six data transmission modes are given in Table 3.6-1. Bit rate, modulation, subcarrier frequencies, and mission phases are indicated for each mode.

A separate PN-modulated subcarrier is transmitted during the lowest-data-rate mode and the emergency mode. This ensures positive synchronization during data recovery.

Planetary science data is processed a five-bit block at a time. Each five-bit block is encoded into a 16-bit biorthogonal code word before transmission.

Table 3.6-1: Telemetry Time Multiplexing And Modulation

MODE	DATA TYPE	BIT RATE	MODULATION	SUBCARRIER FREQUENCY LOWER DATA CHANNEL	SUBCARRIER FREQUENCY UPPER DATA CHANNEL	MISSION PHASES
1	Engrg Capsule	11 1/9) 22 2/9 11 1/9)	Two-Channel Coherent PSK/PM	Data 400 C/S Sync 200 C/S		Launch Acquisition Maneuver
2	Engrg Capsule Cruise Sci.	11 1/9) 11 1/9) 133 1/3 111 1/9)	Coherent PSK/PM	533 1/3 C/S		Cruise
3	Engrg Capsule Stored Engrg	11 1/9) 11 1/9) 133 1/3 111 1/9)	Coherent PSK/PM	533 1/3 C/S		Post Maneuver Option
4	Engrg	5 5/9	Two-Channel Coherent PSK/PM	Data 100 C/S Sync 50 C/S		Emergency Cruise or Encounter
5A	Engrg Cruise Sci. Capsule Planetary Sci.	66 2/3) 166 2/3) 400 166 2/3) 8000	Coherent PSK/PM Lower Coded PSK/PM Upper	1.6 KC	102.4 KC	Encounter and Early Orbital
5B	Engrg Cruise Sci. Capsule Planetary Sci.	66 2/3) 166 2/3) 400 166 2/3) 4000	Coherent PSK/PM Lower Coded PSK/PM Upper	1.6 KC	102.4 KC	Optional Mid- Orbital
5C	Engrg Cruise Sci. Capsule Planetary Sci.	66 2/3) 166 2/3) 400 166 2/3) 2000	Coherent PSK/PM Lower Coded PSK/PM Upper	1.6 KC	102.4 KC	Optional Late Orbital
6	Engrg Cruise Sci. Capsule Planetary Sci.	66 2/3) 166 2/3) 400 166 2/3) 48,000	Coherent PSK/PM Lower Coded PSK/PM Upper	9.6 KC	614.4 KC	Optional Encounter and Orbital

The planetary science data modulates either a 102.4-kilocycle subcarrier (recorded) or 614.4-kilocycle subcarrier (real time) and is the only source of modulation for these subcarriers.

Engineering data, capsule data, relayed capsule data, and cruise science data are time-multiplexed in various modes and provide the modulation for the lower-frequency subcarrier. Only one low-frequency subcarrier is transmitted at a time (see Table 3.6-1). The five lower-frequency subcarrier data formats are given in Figure 3.6-1. A typical Engineering data format is shown in Figure 3.6-2.

Programming of all onboard processors that contribute data to be time-multiplexed onto the lower-frequency subcarrier is synchronized by the timing generator section in the TDSS (see Figure 3.6-3). All data processors are activated for seven-bit periods at the proper time for multiplexing into the lower-subcarrier data stream.

Frame synchronization of the output frame is provided by the TDSS. Each processor provides its own frame synchronization. (Frame synchronization data from the various processors is processed in seven-bit samples, as is all other data.) Frame synchronization on the ground eliminates data bit ambiguity.

3.6.4 On-Board Data Storage

Planetary science data is tape-recorded at 10,000 five-bit blocks per second on command. Readout is automatic at a rate that depends on the operating mode (see Table 3.6-1).

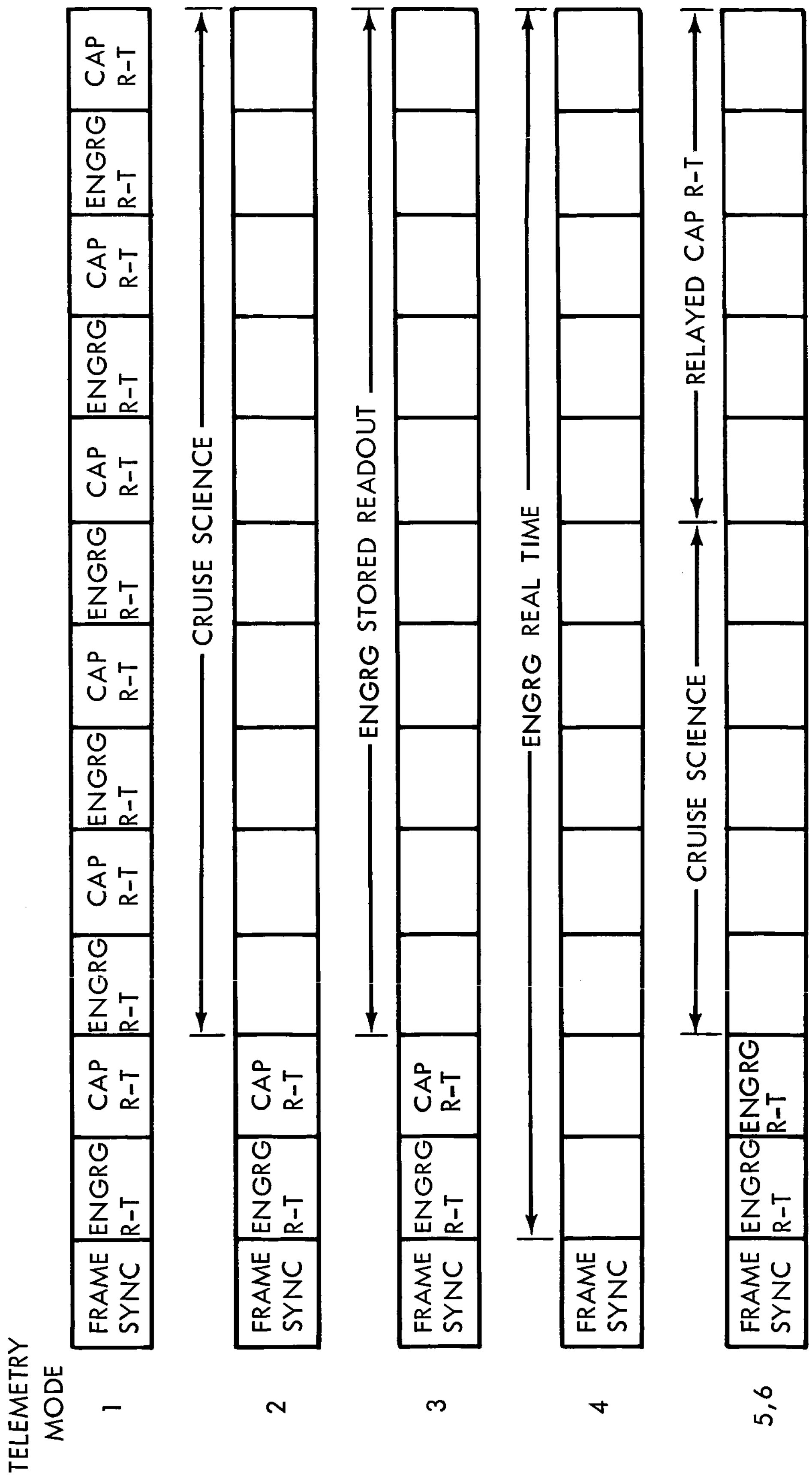


Figure 3.6-1 : Low-Frequency Subcarrier Output Data Formats

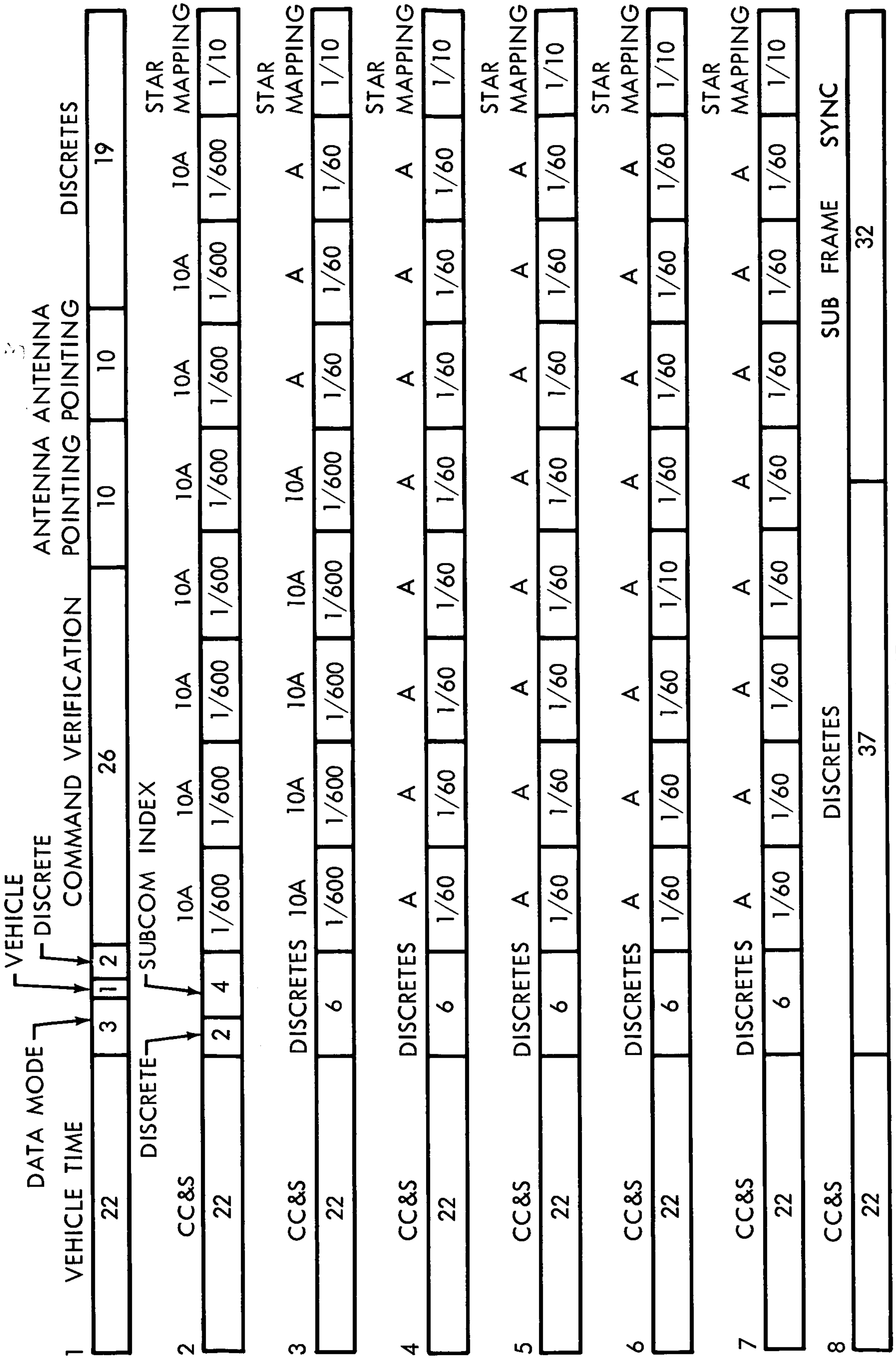
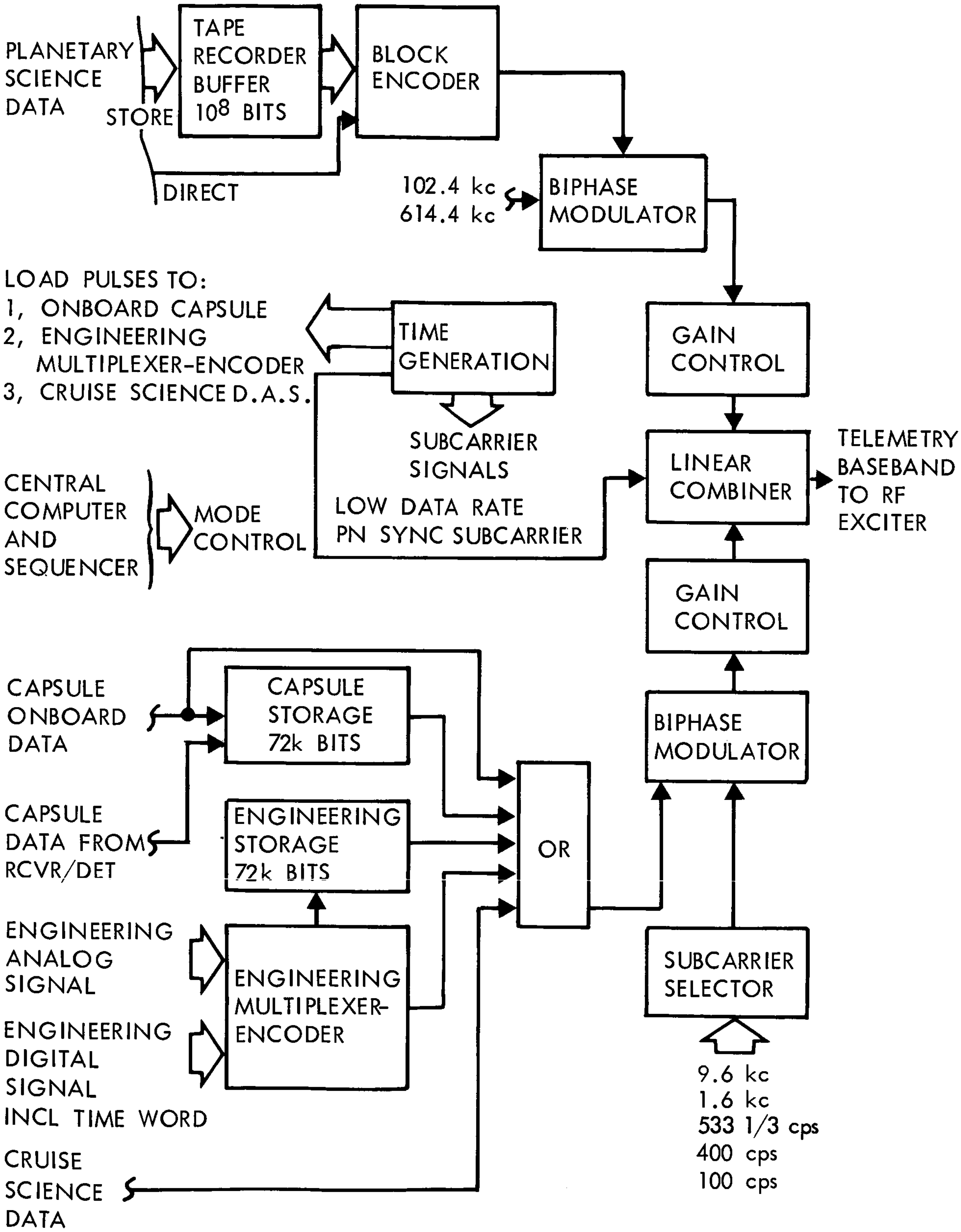


Figure 3.6-2: Typical Bus Engineering Data Format



(Reliability Considerations Not Shown)

Figure 3.6-3: TDSS Block Diagram

Engineering data is stored continuously in the cruise mode with continuous updating. Readout is on command.

Capsule data is stored continuously before capsule launch. Storage is updated continuously. Readout is on command.

Capsule data is stored and read out at appropriate rates following capsule deflection (see Table 3.6-1 Mode 5).

3.6.5 Subcarrier Amplitude Adjustment

Provision is made for adjusting both subcarrier amplitudes by mode commands.

3.7 VOYAGER FLIGHT EQUIPMENT TELEMETRY CHANNEL LIST

3.7.1 Scope

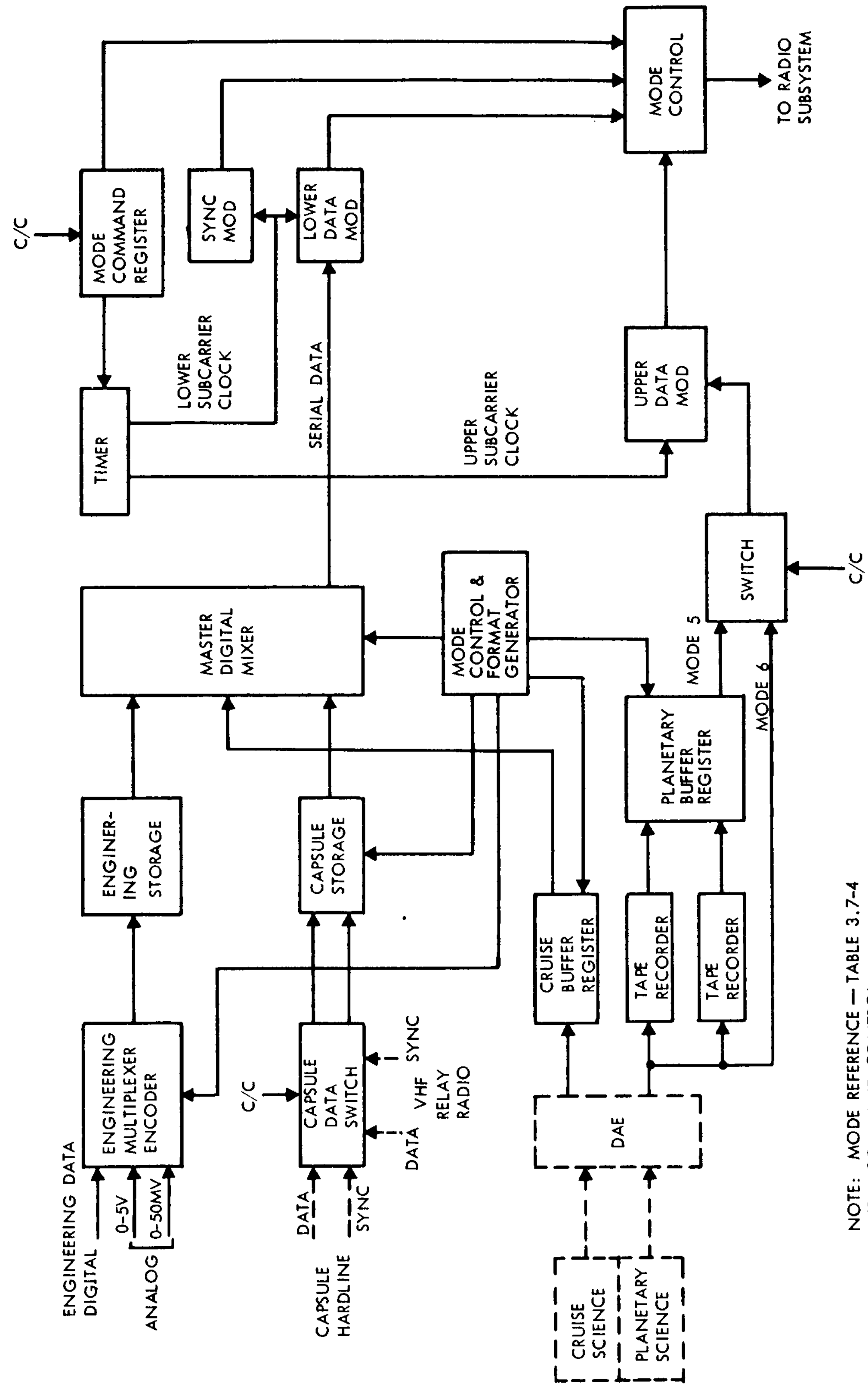
This description provides a preliminary flight data measurements list (Table 3.7-3) of those measurements deemed essential to successful flight operations and analysis of the Voyager spacecraft, including failure modes. In compiling the list, an effort was made to minimize the number and complexity of measurements in each subsystem category, and yet retain the ability to analyze and control spacecraft performance. Engineering measurements on spacecraft performance are listed by subsystem, including the condition and environment of the science package.

The flight data measurements list becomes the "Voyager Flight Equipment Telemetry Channel List" on assignment of format channels for all data items.

3.7.2 Description

3.7.2.1 Commutator (Multiplexer)

Spacecraft engineering signals to the data handling subsystems, both analog and digital, are conditioned with respect to amplitude, pulse width, and format position by an electronic data multiplexer. (See Figure 3.7-1.) The analog channels are then digitized to seven-bit words. Engineering data from the spacecraft science package and the capsule are paralleled with the spacecraft engineering data into a master digital mixer, the function of which is to complete the total signal format consistent with the commanded data mode. Capsule engineering data rate is specified to be ten bits per second.



NOTE: MODE REFERENCE — TABLE 3.7-4
C/C = COMMAND CONTROL

Figure 3.7.1: Voyager Spacecraft Telemetry Data Flow

3.7.2.2 Measurements List

Flight data measurements given herein are tabulated to present pertinent information concerning each measurement. The title of the form includes the subsystem designation. The column headings are:

1) Measurement (name) -- The nomenclature of the measurement generally describes the function.

2) Parameter -- This block lists the units and value range of the measurement.

3) Systems accuracy -- Systems accuracy requirements are listed to provide end instrument, A/D converter, and digital word length criteria.

4) Signal type and bit number -- Analog and digital values given to the telemetry subsystems are listed: (analog, 0-5V, ±2.5V) (dig., 23 bits, 1 bit) etc.

5) Sample rate -- Sample rate is the bits per second rate of sampling any one measurement determined by type of information, impact on flight analysis, proportion of measure with time, and data mode.

Tables--The four tables following are: 3.7-1, Measurements Summary, listing engineering measurements by sample rate, sample type (analog or digital) and bit rate corresponding to each sample rate; 3.7-2, Spacecraft Flight Data, listing measurements per subsystem for sample rate and parametric types; 3.7-3, the Flight Data Measurements List; and 3.7-4, Voyager Telemetry Channel Mode Data, listing modes versus mission phase, data type and subcarrier bit rate.

	SAMPLES SECONDS	ANALOG	DIGITAL	TOTAL	BITS/SEC
Spacecraft Engineering	1/60	39	24	63	8.850
	1/600	30	11	41	0.351
	1/1200	55	22	77	0.514
Spacecraft Science Engineering	1/1200	30		30	0.175
TOTALS		154	57	211	9.89

Table 3.7-1: MEASUREMENTS SUMMARY

Table 3.7-2: Spacecraft Flight Data

	MEASUREMENTS				TYPES OF MEASUREMENTS											
	TOTAL	SAMPLES PER SECOND 1/60	1/600	1/1200	POWER	VOLTAGE	CURRENT	ANGLE	TEMPERATURE	CONDITION	EVENT	STATUS OR MODE	PRESSURE	TIME OR FREQUENCY	POSITION	SIGNAL INTENSITY
ANTENNA	9	2	2	5	X			X	X						X	
RADIO (S/C) + (RELAY)	20	3	6	11	X	X	X	X	X			X				X
TELEMETRY & DATA STORAGE	7	3	-	4					X		X	X	X		X	
ATTITUDE REFERENCE	15	12	3	-				X	X	X						X
AUTOPILOT	12	6	4	2	X			X	X	X		X				
CC&S	17	15	1	1					X		X	X		X		
REACTION CONTROL	14	2	4	8				X	X	X			X			
ELECTRICAL POWER	54	14	14	26		X	X		X	X				X		
THERMAL CONTROL	16	-	-	16				X	X							
PROPULSION	8	6	2	-					X	X			X			
MECHANISMS	9	-	5	4				X	X	X						
SCIENCE PACKAGE ENGINEERING	30	-	-	30		X	X		X							
TOTALS	211	63	41	107												

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	High-Gain Antenna Hinge Angle	Angle	0-360°	$\frac{1}{2}$°	10 bit	1/60
2	High-Gain Antenna Swivel Angle	Angle	0-360°	$\frac{1}{2}$°	10 bit	1/60
3	High-Gain Antenna RF Input	RF Pwr	0-50W	2%	0-50MV	1/600
4	Low-Gain Antenna RF Input	RF Pwr	0-50W	2%	0-50MV	1/600
5	High-Gain Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200
6	Low-Gain Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200
7	High-Gain Antenna Hinge Motor Temp	Temp	±300°F	2%	±2.5V	1/1200
8	High-Gain Antenna Swivel Motor Temp	Temp	±300°F	2%	±2.5V	1/1200
9	VHF Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--ANTENNA

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Receiver AGC (Coarse)	Signal Strength	-90 to -162 dbn	2%	0-5V	1/60
2	Receiver AGC (Fine)	Signal Strength	-125 to -150 dbn	2%	0-5MV	1/600
3	Receiver Static Phase Error	Phase Angle	$\pm 20^{\circ}$	2%	± 2.5V	1/60
4	Receiver L.O. Drive	Osc.Pwr	0-1MW	2%	0-50MV	1/600
5	Exciter Output Power	Osc.Pwr	0-0.2W	2%	0-5V	1/600
6	TWT Helix Voltage	volts	0-2KV	2%	0-5V	1/1200
7	TWT Helix Current	Miliamps	0-20ma	2%	0-5V	1/1200
8	TWT Coll Current	Miliamps	0-100ma	2%	0-5V	1/1200
9	Exciter Voltage #1	volts	0 to -25 volts	2%	± 2.5V	1/1200
10	Exciter Voltage #2	volts	0 to -15 volts	2%	± 2.5V	1/1200
11	Crystal OSC Temp (Oven)	Temp	25 to 150°F	2%	0-5V	1/1200
12	Command Det Mon	Status	yes/no	discrete	1 bit	1/60

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--RADIO

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
13	TWT A Temp	Temp	$\pm 150^{\circ}F$	5%	$\pm 2.5V$	1/1200
14	TWT B Temp	Temp	$\pm 150^{\circ}F$	5%	$\pm 2.5V$	1/1200
15	Receiver A Temp	Temp	$\pm 150^{\circ}F$	5%	$\pm 2.5V$	1/1200
16	Receiver B Temp	Temp	$\pm 150^{\circ}F$	5%	$\pm 2.5V$	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--RADIO (continued)

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Receiver AGC	Signal Strength	-90 to -150 dbn	2%	0-5V	1/600
2	Receiver Static Phase Error	Phase Angle	$\pm 20^{o}$	2%	$\pm$2.5V	1/600
3	Receiver LO Drive	Power	0-1 mw	2%	0-5V	1/600
4	Receiver Temp	Temp	$\pm 150^{o}$F	2%	$\pm$2.5V	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--RELAY RADIO

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Recorder "A" Pressure	x port press	0-4 psia	2%	0-5V	1/1200
2	Recorder "B" Pressure	x port press	0-4 psia	2%	0-5V	1/1200
3	Mode Status	Modes	0-4 psia	discrete	2 bits	1/60
4	Event Counter A	Event	0-100	discrete	7 bits	1/60
5	Event Counter B	Event	0-100	discrete	7 bits	1/60
6	Recorder A Temp	Temp	$\pm 150^{\circ}F$	5%	$\pm 2.5V$	1/1200
7	Recorder B Temp	Temp	$\pm 150^{\circ}F$	5%	$\pm 2.5V$	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--TELEMETRY & DATA STORAGE SYSTEM

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1-6	Star Mapping Signal (6 channels)	Intensity Magnitude	-2.5 to +6	2%	0-5V	1/60
7	Sun Presence Signal	Intensity	yes/no	discrete	1 bit	1/60
8	Canopus Recognition	Intensity	yes/no	discrete	1 bit	1/60
9	Sun Sensor Pitch	Angle	$\pm 5^{\circ}$	2%	± 2.5V	1/60
10	Sun Sensor Yaw	Angle	$\pm 5^{\circ}$	2%	± 2.5V	1/60
11	Canopus Roll	Angle	$\pm 2^{\circ}$	2%	± 2.5V	1/60
12	Canopus Track Power	Cond	yes/no	discrete	1 bit	1/600
13	Canopus Gate	Cond	yes/no	discrete	1 bit	1/600
14	Canopus Sensor Intensity	DN	0-100	2%	0-5V	1/60
15	Canopus Sensor Temp	Temp	$\pm 150^{\circ}$F	2%	± 2.5V	1/600

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--ATTITUDE REFERENCE

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Roll Gyro Error	Angle	$\pm 2^{\circ}$	2%	$\pm$2.5V	1/60
2	Pitch Gyro Error	Angle	$\pm 2^{\circ}$	2%	$\pm$2.5V	1/60
3	Yaw Gyro Error	Angle	$\pm 2^{\circ}$	2%	$\pm$2.5V	1/60
4	Roll Gyro Rate	Angular Rate	$\pm 2^{\circ}$/sec	2%	$\pm$2.5V	1/60
5	Pitch Gyro Rate	Angular Rate	$\pm 2^{\circ}$/sec	2%	$\pm$2.5V	1/60
6	Yaw Gyro Rate	Angular Rate	$\pm 2^{\circ}$/sec	2%	$\pm$2.5V	1/60
7	Accelerometer Power	Cond	yes/no	discrete	1 bit	1/600
8	Roll Spin Motor	Sync Cond	yes/no	discrete	1 bit	1/600
9	Pitch Spin Motor	Sync Cond	yes/no	discrete	1 bit	1/600
10	Yaw Spin Motor	Sync Cond	yes/no	discrete	1 bit	1/600
11	IRU Temp	Temp	$\pm 150^{\circ}$F	5%	$\pm$2.5V	1/1200
12	IRU Power Mode	Cond	yes/no	discrete	1 bit	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--AUTOPILOT

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1-7	Programmer Data	Mode Cond	Instruct Words	Absolute	22 bits	1/60
8	Programmer Parity	Ind	yes/no	discrete	1 bit	1/60
9	Vehicle Time	Time	233 hrs	1/10 sec	22 bits	1/60
10	Command Verify	Verify	Command Word	Absolute	26 bits	1/60
11	A/C Mode Select	Mode Cond	yes/no	discrete	3 bits	1/60
12	Power Mode Select	Cond	yes/no	discrete	1 bit	1/600
13	Unit Temp	Temp	$\pm 150^{0}$	1%	± 2.5V	1/1200
14	Science Data Mode Select	Mode Cond	yes/no	discrete	3 bits	1/60
15	Command Message Identification	Mode Cond	yes/no	discrete	6 bits	1/60
16	Processor #1, #2	Mode Cond	yes/no	discrete	1 bit	1/60
17	Command Decoder #1, #2	Mode Cond	yes/no	discrete	1 bit	1/60

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--CENTRAL COMPUTER & SEQUENCER

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	A/C Tank #1 Pressure	psia	0-375	2%	0-5V	1/60
2	A/C Tank #2 Pressure	psia	0-375	2%	0-5V	1/60
3	Engine Pitch Act + (OIP)	Angle	$\pm 5^{\circ}$	2%	± 2.5V	1/600
4	Engine Pitch Act + (MCP)	Angle	$\pm 5^{\circ}$	2%	± 2.5V	1/600
5	Engine Yaw Act + (OIP)	Angle	$\pm 5^{\circ}$	2%	± 2.5V	1/600
6	Engine Yaw Act + (MCP)	Angle	$\pm 5^{\circ}$	2%	± 2.5V	1/600
7	Jet Driver Pitch +	Cond.	yes/no	discrete	1 bit	1/1200
8	Jet Driver Pitch -	Cond	yes/no	discrete	1 bit	1/1200
9	Jet Driver Yaw +	Cond	yes/no	discrete	1 bit	1/1200
10	Jet Driver Yaw -	Cond	yes/no	discrete	1 bit	1/1200
11	Jet Driver Roll +	Cond	yes/no	discrete	1 bit	1/1200
12	Jet Driver Roll -	Cond	yes/no	discrete	1 bit	1/1200
13	A/C Gas Temp #1	Temp F	$\pm 150^{\circ}$	1%	0-5V	1/1200
14	A/C Gas Temp #2	Temp F	$\pm 150^{\circ}$	1%	0-5V	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST -- REACTION CONTROL

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Solar Panel #1 Current	Amp	0-10A	2%	0-5V	1/60
2	Solar Panel #2 Current	Amp	0-10A	2%	0-5V	1/60
3	Solar Panel #3 Current	Amp	0-10A	2%	0-5V	1/60
4	Battery B1 Voltage	Volts	0-50V	2%	0-5V	1/60
5	Battery B2 Voltage	Volts	0-50V	2%	0-5V	1/60
6	Battery B3 Voltage	Volts	0-50V	2%	0-5V	1/60
7	Battery B1 Discharge	Amp	0-15A	2%	0-5V	1/60
8	Battery B2 Discharge	Amp	0-15A	2%	0-5V	1/60
9	Battery B3 Discharge	Amp	0-15A	2%	0-5V	1/60
10	Battery B1 Charge	Amp	0-5A	2%	0-5V	1/60
11	Battery B2 Charge	Amp	0-5A	2%	0-5V	1/60
12	Battery B3 Charge	Amp	0-5A	2%	0-5V	1/60
13	Unregulated DC Bus Current	Amp	0-30A	2%	0-5V	1/60
14	Unregulated DC Bus Voltage	Volt	0-50V	2%	0-5V	1/60
15	Prime Telecom Regulator	Amp	0-15A	2%	0-5V	1/600

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--ELECTRICAL POWER

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
16	Standby Telecom Regulator	Amp	0-15A	2%	0-5V	1/600
17	Prime Spacecraft Regulator	Amp	0-15A	2%	0-5V	1/600
18	Standby Spacecraft Regulator	Amp	0-15A	2%	0-5V	1/600
19	Regulated DC Bus (S/C)	Volt	0-50V	2%	0-5V	1/600
20	Regulated DC Bus (Telecom)	Volt	0-50V	2%	0-5V	1/600
21	400 cps 1 Ø Bus Voltage	Volt	0-30V	2%	0-5V	1/600
22	400 cps 1 Ø Inverter Current	Amp	0-3A	2%	0-5V	1/600
23	400 cps 3 Ø Bus Voltage	Volt	0-30V	2%	0-5V	1/600
24	400 cps 3 Ø Inverter Current	Amp	0-3A	2%	0-5V	1/600
25	2400 cps Inverter Prime	Amp	0-5A	2%	0-5V	1/600
26	2400 cps Inverter Standby	Amp	0-5A	2%	0-5V	1/600
27	Power Sync Frequency	cycles/ second	350 to 450 cps	2%	0-5V	1/600
28	Power Sync Frequency	kc/sec	2.2 to 2.6 kc	2%	0-5V	1/600
29	Solar Panel #1 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
30	Solar Panel #2 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--
ELECTRICAL POWER (Continued)

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
31	Solar Panel #3 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
32	Regulator Output Ripple	mili-volts	0-500MV	2%	0-5V	1/1200
33	Battery B1 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
34	Battery B2 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
35	Battery B3 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
36	Telecom Regulator #1 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
37	Telecom Regulator #2 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
38	Spacecraft Regulator #1 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
39	Spacecraft Regulator #2 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
40	PS&L Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
41	2.4 kc Inverter #1 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
42	2.4 kc Inverter #2 Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
43	400 cps 1 Ø Inverter Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
44	400 cps 3 Ø Inverter Temp	^{o}F	$\pm 150^{o}$	5%	$\pm 2.5V$	1/1200
45-54	Relay RL-1 thru RL-10 Status	Cond	yes/no	discrete	10 bits Total	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--
ELECTRICAL POWER (Continued)

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Louver 1 Position	Angle	$0-90^\circ$	2%	0-5V	1/1200
2	Louver 2 Position	Angle	$0-90^\circ$	2%	0-5V	1/1200
3	Louver 3 Position	Angle	$0-90^\circ$	2%	0-5V	1/1200
4	Louver 4 Position	Angle	$0-90^\circ$	2%	0-5V	1/1200
5	Louver 5 Position	Angle	$0-90^\circ$	2%	0-5V	1/1200
6	Louver 6 Position	Angle	$0-90^\circ$	2%	0-5V	1/1200
7	Coldplate 1 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
8	Coldplate 2 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
9	Coldplate 3 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
10	Coldplate 4 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
11	Coldplate 5 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
12	Coldplate 6 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
13	Coldplate 7 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
14	Coldplate 8 Temp	Temp	$\pm 150^\circ$	5%	$\pm 2.5V$	1/1200
15	M/C Motor Shield Temp	Temp	$0-1K^\circ F$	5%	0-5V	1/1200
16	O/1 Motor Shield Temp	Temp	$0-1K^\circ K$	5%	0-5V	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--THERMAL CONTROL

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Gas Supply Pressure	psia	0-4000	5%	0-5V	1/60
2	Fuel Tank Pressure	psia	0-400	5%	0-5V	1/60
3	Gas Supply Temperature I	^{o}R	300-600	5%	0-5V	1/60
4	Gas Supply Temperature II	^{o}R	300-600	5%	0-5V	1/60
5	Fuel Tank Temperature I	^{o}R	400-600	5%	0-5V	1/600
6	Fuel Tank Temperature II	^{o}R	400-600	5%	0-5V	1/600
7	Motor Temperature	^{o}R	400-1500	5%	0-5V	1/60
8	Engine Valve Driver-Fuel	Cond	yes/no	Discrete	1 bit	1/60

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--PROPULSION

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Solar Panel #1 Deploy	Cond	yes/no	discrete	1 bit	1/600
2	Solar Panel #2 Deploy	Cond	yes/no	discrete	1 bit	1/600
3	Solar Panel #3 Deploy	Cond	yes/no	discrete	1 bit	1/600
4	Magnetometer Boom Deploy	Cond	yes/no	discrete	1 bit	1/600
5	Scan Platform Position	Angular Position	$0-90^{\circ}$	2%	0-5V	1/600
6	Scan Platform Deploy	Cond	yes/no	discrete	1 bit	1/1200
7	Optics Cover Position	Cond	yes/no	discrete	1 bit	1/1200
8	Gimbal Motor 1 Temp	Temp	$\pm 150^{\circ}$F	2%	± 2.5V	1/1200
9	Gimbal Motor 2 Temp	Temp	$\pm 150^{\circ}$F	2%	± 2.5V	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--MECHANISMS

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1-18	Science Equipment Temperatures	Temp	-----	-----	-----	1/1200
19-24	Science Equipment Voltages	Volts	-----	-----	-----	1/1200
25-30	Science Equipment Currents	Amps	-----	-----	-----	1/1200

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--SCIENCE PACKAGE ENGINEERING

IDEN.	MEASUREMENT	PARAMETER		REQUIRED ACCURACY	SIGNAL AN/DIG	RATE (SAMPLES/SECOND)
		UNIT	RANGE			
1	Battery Temp	Temp "F"	$\pm 150^{o}F$	2%	$\pm 2.5V$	1/60
2	Battery Voltage	Volt	0-50V	2%	0-5V	1/60
3	Umbilical Connect	Cond	yes/no	discrete		1/60
4	Clock Freq	cps	± 5 cps	2%	$\pm 2.5V$	1/60
5	CC&S Mode Select	Cond	yes/no	discrete	6 bits	1/60
6	Clock Freq (Sync)	Cond	yes/no	discrete	1 bit	1/60
7	Fuel Tank #1 Pressure	psia	0-3K	2%	0-5V	1/60
8	Fuel Tank #2 Pressure	psia	0-3K	2%	0-5V	1/60
9	A/C Gas Tank Pressure	psia	0-3K	2%	0-5V	1/60
10	A/C Gas Tank Temp	Temp	$\pm 150^{o}F$	2%	$\pm 2.5V$	1/60
11	Separation Sequence Select	Cond	yes/no	discrete	1 bit	1/60
12	Pyro Activate	Cond	yes/no	discrete	1 bit	1/60
13	Fuel Tank Temp	Temp "F"	$\pm 150^{o}F$	2%	$\pm 2.5V$	1/60
14	Oxidizer Tank Temp	Temp "F"	$\pm 150^{o}F$	2%	$\pm 2.5V$	1/60
15	RTG - Radiator 1 Temp	Temp "F"	$\pm 150^{o}F$	2%	$\pm 2.5V$	1/60
16	RTG - Radiator 2 Temp	Temp "F"	$\pm 150^{o}F$	2%	$\pm 2.5V$	1/60

Table 3.7-3: FLIGHT DATA MEASUREMENT LIST--CRUISE CAPSULE DATA (TYPICAL)

Table 3.7-4: Voyager Telemetry Channel Mode Data

MODE	MISSION PHASE	TELEMETRY CHANNEL DATA TYPE & RATES	BIT RATE SUBCARRIER	
			UPPER	LOWER
I	Launch, Acquisition, Cruise Maneuvers	S/C Engr @ 11-1/9 bps Capsule Eng. @ 11-1/9 bps		22-2/9
II or III	Cruise and Post Maneuver Option	Same as I plus Cruise Science or Stored S/C Engr @ 111-1/9 bps		133-1/3
IV	Emergency Cruise or Encounter	S/C Engr @ 5-5/9 bps		5-5/9
VA	Primary Encounter and Orbital	S/C Engr @ 66-2/3 bps Cruise Science @ 166-2/3 bps Capsule Data @ 166-2/3 bps Planetary Science @ 8000 bps	8000	400
VB	Optional Late Orbital	Same as VA except Planetary Science @ 4000 bps	4000	400
VC	Optional Late Orbital	Same as VA except Planetary Science @ 2000 bps	2000	400
VI	Optional Encounter and Orbital	Same as VA except Planetary Science @ 48,000 bps	48,000	400

3.8 GUIDANCE AND NAVIGATION MANEUVER ERROR ALLOCATIONS

3.8.1 Pointing Angle Accuracy

To ensure sufficient accuracy of midcourse velocity corrections, it is necessary that the thrust vector be oriented within a given small tolerance of a specific, precomputed line in inertial space along which the velocity correction is to be made. This tolerance has been set at a 3-sigma limit of 1.72 degrees, which essentially means that under no circumstances can the thrust vector fall outside a cone of 3.44 degrees included angle.

Thrust-vector pointing error results from five sources (listed in Table 3.8-1): inertial and optical sensors, initial conditions of the vehicle, errors introduced when actually performing the maneuver, and thrust-vector-control center-of-gravity offset. Since the sensors provide the orientation reference for the vehicle, they must be closely aligned with the vehicle axes. The gyros are used as both rate- and position-measuring devices. Thus, their sensitivity (scale factor) and undesired drift characteristics enter into the error considerations. Nonorthogonality of their input axes produces errors due to cross-axis coupling during maneuvers. The optical sensors are subject to null offsets, which refer to the relation between zero output voltage and zero angle. Since the vehicle is a torque-free body within specified error limits, it undergoes sustained oscillations between these limits. Because these errors are variable, they are added directly at their maximum levels, which occur twice per limit cycle. Finally, installation of the main engine introduces errors since its null position is not precisely aligned with the vehicle center of gravity. The nozzle itself expels the gasses slightly offset from the nozzle centerline.

Table 3.8-1 also shows the manner in which the 1.72-degree error allowance is budgeted between the five main error sources. The allowance is weighted in such a way as to give the greatest tolerance to the thrust vector control in the third midcourse maneuver since this, after all, is the prime mover in establishing the velocity correction.

3.8.2 ΔV Measurement Accuracy

The three-sigma allowable error in the magnitude of the velocity increment is 3 percent. This error can be allocated to two areas, the measurement error and the propulsion system error. Since the midcourse correction and orbit-insertion propulsion systems are different, it is convenient to treat each maneuver separately.

Midcourse ΔV--The magnitude of the midcourse ΔV corrections will lie between 0.1 and 75 meters per second. The acceleration level will be approximately 0.013 g's. The 3-sigma error in midcourse ΔV magnitude is as follows:

Accelerometer Null Bias = $100 \times \frac{3 \times 10^{-4} g}{0.013\ g} = 2.3\%$

Accelerometer Scale Factor Error = 0.3%

Engine Tail-off Uncertainty = 0.005 m/sec.

Accelerometer Resolution Error = 0.01 m/sec.

For most midcourse ΔV maneuvers, the accuracy is seen to be limited by the accelerometer bias. One-sigma ΔV magnitude accuracy is 0.8 percent.

Table

Error Source	Error Description Breakdown		
Inertial Sensors (G10 Gyros)	Mechanical Misalignment of Sensitive Axes with Vehicle:		
	Drift During Inertial Hold: *		Random (0.015°/hr)
			Bias (0.15°/hr)
			Temp. (0.005°/hr/°F)
	Drift During Thrust: **	g - Sensitive	0.2°/hr/g
			0.5°/hr/g
		g^2 - Sensitive	9.6°/hr/g^2
			1.6°/hr/g^2
	Gyro Nonorthogonality: +		
Optical Sensors (Barnes JPL Ball Bros.)	Mechanical Misalignment of Sensitive Axis with Vehicle:		
			Sun Sensor
			Canopus Sensor
	Null Offset (Mechanical, Internal, and Electrical):		
			Sun Sensor
			Canopus Sensor
	Sensing Accuracy:		
			Sun Sensor
			Canopus Sensor
Initial Condition	Limit-Cycle Attitude:		
	Switching Amplifier Null Offset:		
Maneuver Control	Gyro Torquer Scale Factor (330 ppm): +		
	CC&S Control Quantization:		
Thrust/ CG Alignm't	Thrust Vector Control (Autopilot) ++		
	Vehicle Angular Error (Without Thrust Misalignment)		
	Total Thrust Vector Angular Error		

*Based on 100-Minute Third Midcourse Correction and 70-Minute Orb
10°F Temp Change

**2-Minute, 2-g Thrust for Insertion

+ 100-Degree Maneuvers, Pitch and Roll

++ Includes Effects of Engine Installation and Thrust Tolerances

3.8-1: Vehicle and Thrust Vector Angular Errors at Maneuvers

Third Midcourse Maneuver (3σ degrees)			Orbit Insertion (3σ degrees)			Midcourse Error Totals (3σ) (degrees)	Insertion Error Totals (3σ) (degrees)	Error Budget Allocation (3σ) (degrees)
Pitch	Yaw	Roll	Pitch	Yaw	Roll			
0.05	0.05	0.05	0.05	0.05	0.05			
0.025	0.025	0.025	0.018	0.018	0.018			
0.250	0.250	0.250	0.180	0.180	0.180			
0.08	0.08	0.08	0.06	0.06	0.06			
					0.033			
			0.013	0.013	0.013			
			0.002	0.002				
0.10	0.08	0.10	0.10	0.08	0.10	0.463	0.340	0.500
0.08	0.08	0	0.11	0.11	0			
0	0	0.04	0	0	0.04			
0.025	0.025	0	0.032	0.032	0			
0	0	0.025	0	0	0.025			
0.005	0.005	0	0.007	0.007	0			
0	0	0.10	0	0	0.10	0.162	0.196	0.300
0.15	0.15	0.15	0.20	0.20	0.20			
0.03	0.03	0.03	0.04	0.04	0.03	0.265	0.354	0.450
0.033	0	0.033	0.033	0	0.033			
0.01	0	0.01	0.01	0	0.01	0.049	0.049	0.200
1.400	0.230	0	0.600	0.600	0	1.420	0.848	1.540
0.281	0.273	0.317	0.295	0.277	0.295	0.560	0.530	0.765
1.430	0.386	0.317	0.668	0.662	0.295	1.540	1.000	1.720

· Injection Maneuvers,

<u>Insertion ΔV</u>--The insertion ΔV will be provided by a fixed-impulse solid engine, with secondary injection thrust vector control. The error in the insertion ΔV magnitude will be due to the following sources:

1) Uncertainty in propellant weight;
2) Variation in fuel specific impulse;
3) Variation in inert mass.

The 3-sigma combined effect of variations in fuel weight and fuel specific impulse is 0.6 percent or 30 feet per second for a 5000-foot-per-second ΔV.

The variation in inert mass is due to error in initial vehicle weight and uncertainty in the amount of midcourse fuel and reaction-control fuel consumed up to the time of injection. Also included is the uncertainty in the amount of secondary-injection fuel used during insertion.

Weighing error = $\left(\frac{0.05\%}{100} \times 2000 \text{ pounds}\right)$ =	1 pound
Uncertainty in weight of reaction fuel and midcourse fuel at insertion	6
Uncertainty in weight of secondary injection fuel used-equivalent inert weight at start burn	4
TOTAL	11 pounds

The ΔV error corresponding to an error in inert weight of 11 pounds is 33 feet per second.

Three sigma ΔV error = $\left[30^2 + 33^2\right]^{\frac{1}{2}}$ = 44 feet per second, or 0.88 percent of the velocity magnitude. The 1 sigma velocity error will then be 0.29 percent.

3.9 FUNCTIONAL DESCRIPTION VOYAGER FLIGHT EQUIPMENT FLIGHT SEQUENCE

3.9.1 Scope

This description covers the nominal sequence of operations performed by the spacecraft from the period immediately preceding launch until completion of the mission. A logical and detailed analysis of this sequence information provides a basis for determining many CC&S functions, subsystem modes and direct command requirements. The flight sequence charts are used to examine the entire decision making process for the mission. Special attention is given to the functions which optimize mission success to identify problems requiring design or mission planning attention.

3.9.2 Flight Sequence

3.9.2.1 General Sequence of Events

The nominal flight sequence of operations for the 1971 Voyager represents an expansion of mission profile data provided by JPL. The nominal sequence for the Flight Spacecraft is summarized on the "Flight Sequence Subsystem Correlation Chart", Figure 3.9-2. This chart was prepared using the "Project Elements Per Flight Phase Chart", Figure 3.9-1 as a guideline. The horizontal lines on the correlation chart represent the 14 phases of the mission as contained in the mission profile. (Page 7 Preliminary Voyager 1971 Mission Specification). The vertical portions of the chart represent the Flight Spacecraft subsystems and their functions per flight phase.

A general discussion of the nominal sequence follows. The detailed sequence of events is described in 3.9.3.2. Any event or sequence of

	PRELAUNCH AT ETR	LAUNCH AND INJECTION	ACQUISITI
LAUNCH VEHICLE SYSTEM	System checkout, mating, and combined systems testing.	Temperature control for spacecraft. Telemetry from overall spacecraft. Shroud separation. Orientation in parking orbit through spacecraft separation. Assurance that Launch Vehicle meets contamination requirements.	
SPACECRAFT SYSTEM	System checkout. Spacecraft — capsule mating and combination testing, fuel and pyro at explosive safe facility. Mate to adaptor and shroud, transport to pad, launch-vehicle mate and testing. Power to subsystems and capsule.	Temperature control after shroud off. Telemetry to launch vehicle. Separate from launch vehicle. Make engineering measurements. Environment control. Power to subsystems and capsule.	Acquire referenc objects. Provid power to subsyst and capsule. M capsule. Enviro tal control. Pro rf signal to Earth Make engineerin measurements.
CAPSULE SYSTEM	Final checkout, propulsion, radioisotope thermoelectric generator fuel and pyro installation and final sterilization at explosive safe facility. Spacecraft — capsule mate (see Spacecraft System).	Telemetry to spacecraft, temperature control after shroud off. Make engineering measure	
MISSION OPERATIONS SYSTEM	Operational readiness test. Spacecraft and Deep-Space Network compatibility test.	Monitor telemetry. Receive and evaluate Eastern Test Range prediction.	Monitor telemetr Send backup com mands. Send De Space Network station predictio
LAUNCH OPERATIONS SYSTEM	Scheduling and coordination of Eastern Test Range activities.	Control launch. Monitor telemetry. Provide tracking to injection.	
DEEP SPACE NETWORK	Checkout and training Support Mission Operations System.	Provide tracking as required. Support Mission Operations System.	Provide two-way

SOURCE: JPL Preliminary Voyager 1971 Mission Specification, dated 1 May 1965, Page

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ON	INTERPLANETARY CRUISE	INTERPLANETARY TRAJECTORY CORRECTION	SPACECRAFT-CAPSULE SEPARATION AND DEFLECTION	CAPSULE CRUISE	
e : ems onitor nmen- vide . ng	Attitude control power to subsystems and capsule. Monitor capsule status. Provide rf signal to Earth. Make interplanetary scientific and engineering measurements. Environment control.	Provide power to subsystems and capsule. Monitor capsule. Orient thrust vector. Provide $\triangle$ V. Temperature control. Make engineering measurements. Provide rf signal to Earth. Reacquisition.	Provide preseparation commands to capsule. Provide telemetry relay from capsule. Orient separation vector. Separate sterilization canister and maintain contamination requirements. Maintain temperature control. Make engineering measurements. Provide rf signal to Earth.	Provide telemetry relay Spacecraft power to spa tems. Temperature con to Earth. Make science measurements. Environ	
ments.			Separate from spacecraft. Capsule attitude control. Provide $\triangle$ V. Provide power, temperature control and sequencing. Make engineering measurements. Provide rf signal to spacecraft.	Separate unessential equipment. Capsule attitude control. Provide power, temperature control, and sequencing. Make engineering measurements. Provide rf signals to spacecraft.	S a S d D re p c ir a a rf
y. n- ep n.	Monitor telemetry and control overall spacecraft as necessary. Determine orbit characteristics.	Monitor telemetry Determine and send midcourse command. Control spacecraft as necessary. Determine orbit characteristics.	Monitor telemetry. Determine and send separation and deflection commands. Determine orbit characteristics.	Monitor telemetry from spac Determine orbit characterist	
communication lock. Provide tracking support (Mission Operations System)				Possible direct link communications.	

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CAPSULE ENTRY AND DESCENT	LANDING AND OPERATIONS	ORBITAL INSERTION	ORBITAL OPERATIONS
attitude control. cecraft subsys- rol. Rf signal and engineering nental control. See Figure 3.9-2 for Expansion	Provide relay if utilized	Orient retrothrust vector. Confirm attitude to Earth. Provide ΔV. Provide power. Temperature control. Make engineering measurements. Provide rf signal. Reacquisition.	Orient planetary instrumentation. Make science and engineering measurements. Store and read out data. Attitude control. Provide power, temperature control, rf signal and environmental control.
abilize to low ngle of attack. urvive heat and eceleration pulse. eploy subsonic tarder. Provide ower, temperature ontrol, and sequenc- g. Measure temper- ure, pressure, and cceleration. Provide signal to spacecraft.	Absorb landing impact Separate retarder. Erect S-band antenna. Provide power, temperature control, and sequencing. Locate and orient science payload. Make science and engineering measurements. Provide rf signal to Earth. Provide relay if utilized.		
ecraft. ics.	Monitor telemetry from spacecraft and capsule. Determine landing site.	Monitor telemetry. Determine and send commands. Determine orbit characteristics.	Monitor telemetry and control spacecraft as necessary. Determine orbit characteristics.
Provide two-way communication lock. Provide tracking support (Mission Operations System)			

Figure 3.9-1: Project Element Functions per Flight Phase

1.0 PRELAUNCH AT ETR → 2.0 LAUNCH & INJECTION (INCLUDES COUNTDOWN) →

	1.0 PRELAUNCH AT ETR	2.0 LAUNCH & INJECTION (INCLUDES COUNTDOWN)
GROUND COMPLEX (MOS, LOS, & DSN)	OPERATIONAL READINESS TEST, S/C-DSN COMPATIBILITY TEST SCHED & COORD ETR ACTIVITIES C/O & SUPPORT MOS (DSN ONLY) LOAD CC&S WITH FLIGHT PROGRAM	MOS/LOS A) COMMAND LIFTOFF B) TRACK VEHICLE DURING BOOST C) SUPPLY FLIGHT COMMANDS (AS REQ) D) RECEIVE & ANALYZE DATA FROM S/C & BOOSTER DSN E) STANDBY ON ALERT F) COMMUNICATE WITH ETR CHANGE FROM MOS/LOS TO DSN C ON INJECTION INTO TRANSMARS A) PROVIDE SFOF/DSIF WITH AN DSN SEARCH DATA B) RECEIVE ANTENNA SEARCH DA C) SEARCH FOR & ACQUIRE S/C D) ESTABLISH & VERIFY CONTRO E) TRACK S/C (1-WAY) F) RECEIVE & ANALYZE ENGRG D
SPACECRAFT TELECOMMUNICATIONS	SUBSYSTEM C/O & STATUS MONITORING	A) TRANSMIT ENGINEERING DATA VIA CENTAUR TELEMETRY B) TRANSMIT ENGINEERING DATA VIA LOW POWER LAUNCH EXCITER C) RECEIVE POWER FROM E/P
CENTRAL COMPUTER & SEQUENCER (CC&S)	A) SUBSYSTEM C/O & STATUS B) COMMAND OTHER SUBSYSTEMS FOR C/O & STATUS MONITORING C) READY ALL SUBSYSTEMS FOR LAUNCH D) LOAD CC&S WITH FLIGHT PROGRAM	PROVIDE BACKUP COMMANDS AS REQUIRED
ATTITUDE REFERENCE SUBSYSTEM AUTOPILOT SUBSYSTEM REACTION CONTROL SUBSYSTEM (RCS)	SUBSYSTEM C/O & STATUS MONITORING	A) ATTITUDE REFERENCE — GYROS OFF DURING LAUNCH B) AUTOPILOT — OFF C) RCS — OFF
MIDCOURSE CORRECTION PROPULSION SYSTEM ORBIT INJECTION PROPULSION SYSTEM	SUBSYSTEM C/O & STATUS MONITORING	
ELECTRICAL POWER SUBSYSTEM	SUBSYSTEM C/O & STATUS MONITORING	A) PROVIDE ENGRG DATA FOR TELECOMUNICATIONS SUBSYSTEM B) PROVIDE BATTERY POWER TO: • TELECOMMUNICATIONS SUBSYSTEM • CC&S • ATTITUDE REFERENCE SUBSYSTEM • AUTOPILOT SUBSYSTEM
S/C STRUCTURE SUBSYSTEM S/C MECHANISMS SUBSYSTEM INSTALLATION CABLES & TUBING	SUBSYSTEM C/O & STATUS MONITORING	A) PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT B) PROVIDE ATTACHMENT FOR CAPSULE C) SUPPORT FLT CAPSULE
TEMPERATURE CONTROL SUBSYSTEM	SUBSYSTEM C/O & STATUS MONITORING SPACECRAFT COOLING SUPPLIED BY CENTAUR	A) PROVIDE HEAT SINK COOLING CAPABILITY UP TO SHROUD JETTISON B) TEMPERATURE CONTROL AFTER SHROUD JETTISON
PYROTECHNIC SUBSYSTEM	SUBSYSTEM C/O & STATUS MONITORING	
SCIENCE EXPERIMENTS (GFE)	SUBSYSTEM C/O & STATUS MONITORING S/C MAGNETIC MAPPING (FAT)	
DATA AUTOMATION EQUIPMENT (GFE)		

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REPEAT UP TO

→ **3.0 ACQUISITION** → **4.0 INTERPLANETARY CRUISE** →

	3.0 ACQUISITION	4.0 INTERPLANETARY CRUISE	
NTROL RBIT ENNA TA . OF S/C TA	DSN A) MONITOR BOOSTER SEPARATION B) MONITOR SOLAR PANEL DEPLOYMENT C) MONITOR ANTENNA DEPLOYMENT D) PROVIDE COMMANDS TO BACK UP CC&S AS REQUIRED E) TRACK S/C F) REC DATA G) MONITOR SCIENCE DEPLOYMENT DSN H) MONITOR & VERIFY ACQUISITION OF SUN I) MONITOR & VERIFY ACQUISITION OF CANOPUS J) COMMAND REACQUISITION OF CANOPUS (AS REQD) K) MONITOR S/C TRAJECTORY L) UPDATE CC&S TRAJECTORY PARAMETERS FOR PITCH, YAW, & ROLL M) COMPUTE CC&S PITCH,YAW,&ROLL POLARITY	DSN A) TRACK S/C B) RECEIVE & DISPLAY SCIENCE & ENGINEERING DATA C) MONITOR S/C & CAPSULE STATUS D) PROCESS DATA ON EARTH TO OBTAIN GUIDANCE COMMANDS (STORED COMMANDS & START TIMES)	A) B) C) D) E) F)
	A) TRANSMIT ENGINEERING DATA B) RECEIVE POWER FROM E/P C) RECEIVE VIA LOW-POWER LAUNCH EXCITER DETECT & SEND TO CC&S COMMAND SIGNALS FROM EARTH D) TRANSMIT CELESTIAL REFERENCE ACQUISITION TO DSN E) TRANSMIT VERIFICATION OF DEPLOYMENT OF SOLAR PANELS, ANTENNAS, ETC TO EARTH F) TWO-WAY TRACKING	A) TRANSMIT ENGINEERING & SCIENCE DATA VIA DATA MODE NO. 2 (OPTIONAL MODE NO. 3) B) TRANSMIT CAPSULE ENGINEERING DATA C) TRANSMIT CONE-ANGLE SETTINGS (CANOPUS TO EARTH) D) EXERCISE & CALIBRATE HIGH-GAIN ANTENNA T + 40 DAYS E) SWITCH FROM LOW-GAIN TO HIGH-GAIN ANTENNA T + 80 DAYS F) PROVIDE RANGING SIGNAL TO A MAX. RANGE OF 8.0 X 10^6 KM (NOMINAL) G) SWITCH TRANSMISSION FROM LAUNCH EXCITER TO TWT POWER AMPLIFIER AND DATA MODE NO. 2	A) B) C) D) E) F) G)
	A) COMMAND SIGNALS TO PYROTECHNICS & MECH TO DEPLOY • SOLAR PANELS • HIGH-GAIN ANTENNA • VHF ANTENNA • LOW-GAIN ANTENNA • SCIENCE BOOM B) SWITCH ON GYROS & SELECT MODES C) ENABLE SUN-SENSOR ROLL AND PITCH CONTROL D) RECEIVE SUN PRESENCE SIGNAL E) TURN ON CANOPUS TRACKER F) SWITCH ROLL CONTROL TO CANOPUS SENSOR G) ACTIVATE MAGNETOMETER (CALIBRATION ROLL) H) RECEIVE CANOPUS PRESENCE OUTPUT SIGNAL I) TRANSMIT VERIFICATION TO TELECOMMUNICATION SUBSYSTEM (CANOPUS & SUN PRESENCE) J) PERFORM COMMAND FUNCTIONS FOR CANOPUS OVERRIDE AS REQUIRED K) BACKUP COMMANDS AS REQUIRED	A) COMMAND TELECOMMUNICATION TO DATA MODE NO. 2 B) COMMAND CRUISE SCIENCE ON (WARMUP) — SCIENCE INSTRUMENTS C) COMMAND DATA RECORDERS ON D) COMMAND TWTA ON E) SWITCH CANOPUS ANGLE AS REQUIRED F) INITIATE CRUISE SCIENCE DATA ACQUISITION G) COMMAND TELECOMMUNICATION CHANGE FROM OMNI TO HIGH-GAIN ANTENNA H) UPDATE HIGH-GAIN ANTENNA POSITION (AS REQUIRED) I) COMMAND TELECOMMUNICATIONS — CHANGE DATA TRANSMISSION RATES J) COMMAND RECALIBRATION OF SCIENCE ELEMENTS AS REQUIRED K) BACKUP COMMANDS AS REQUIRED	A) B) C) D) E) F) G) H) I) J) K) L) M) N)
	A) RECEIVE CC&S COMMAND SWITCH ON GYROS, RCS, & AUTOPILOT B) DAMP ROTATION C) YAW & PITCH SPACECRAFT TO ACQUIRE SUN D) RELAY SUN ACQUISITION SIGNAL TO CC&S E) TURN ON CANOPUS SENSOR, ROLL 360 TO CALIBRATE MAGNETOMETER & PROVIDE STAR MAP F) ROLL TO ACQUIRE CANOPUS G) RELAY ACQUISITION SIGNAL (CANOPUS) TO CC&S H) PERFORM CANOPUS OVERRIDE ROLL MANEUVER AS REQUIRED	A) UPDATE CANOPUS CONE ANGLE ON COMMAND B) SWITCH AUTOPILOT TO CRUISE MODE C) MAINTAIN S/C ATTITUDE TO CELESTIAL REFERENCES DURING CRUISE	A) B) C) D) E) F) G)
			M A) B) C) D) E) F)
	A) PROVIDE POWER TO PYROTECHNIC SUBSYSTEMS FOR SQUIB FIRINGS B) PROVIDE POWER TO MECHANISMS C) ACTIVATE SOLAR POWER SYSTEM AFTER SUN ACQUISITION (AUTOMATIC) D) TRANSMIT "VOLTAGE SATISFACTORY" SIGNAL TO CC&S E) PROVIDE POWER TO: • TELEMETRY • CC&S • TEMPERATURE CONTROL • AUTOPILOT	A) PROVIDE CONDITIONED SOLAR ELECTRICAL POWER TO: • TELEMETRY • CC&S • ATTITUDE REF • AUTOPILOT • TEMPERATURE CONTROL • SCIENCE • CAPSULE B) CHARGE BATTERIES	A)
	A) PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT B) DRIVE SOLAR PANELS TO LIMIT STOPS C) PROVIDE OUT & LOCK SIGNALS TO TELEMETRY D) DRIVE HIGH-GAIN ANTENNA TO OPERATING POSITION & LOCK E) DRIVE VHF & OMNI ANTENNAS TO OPERATING POSITION & LOCK F) DRIVE MAGNETOMETER BOOM TO OPERATING POSITION & LOCK G) SUPPORT FLT CAPSULE	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D) PROVIDE ACCEPTABLE STATIC & DYNAMIC LOAD ENVIRONMENTS E) SUPPORT FLIGHT CAPSULE	A) B) C) D) E)
	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL	PR
	RECEIVE SQUIB FIRING SIGNALS FOR DEPLOYMENT OF: • SOLAR PANEL • MAGNETOMETER BOOM • LOW-GAIN ANTENNA • HIGH-GAIN ANTENNA • VHF ANTENNA		A) B) C)
	A) CALIBRATE MAGNETOMETER DURING S/C ROLL B) CALIBRATE OTHER SCIENCE INSTR AS REQUIRED	A) RECEIVE CRUISE SCIENCE "ON" COMMAND B) ACQUIRE CRUISE SCIENCE DATA C) TRANSFER CONDITIONED DATA TO DAE D) RECEIVE POWER FROM E/P E) RECALIBRATE SCIENCE EXPERIMENTS AS REQUIRED	A) B)
		A) RECEIVE DAE "ON" COMMAND B) PROCESS & TRANSFER DATA TO TM	

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OUR TIMES

5.0 INTERPLANETARY TRAJECTORY CORRECTION

6.0 S/C CAPSULE SEPARATION

7.0 FLIGHT
TRAJEC
DEFLEC

ROVIDE ROLL TURN MAGNITUDE & DIRECTION ROVIDE YAW TURN MAGNITUDE & DIRECTION ROVIDE MIDCOURSE MOTOR BURN DURATION AND ΔV ROVIDE INITIATE MANEUVER SEQUENCE COMMAND ECEIVE VERIFICATION OF CORRECTION MANEUVER DEGREES OF YAW & ROLL THRUST DURATION REACQUISITION OF CELESTIAL REFERENCES RACK SPACECRAFT	A) INITIATE PRESEPARATION SEQUENCE B) RECEIVE CAPSULE STATUS DATA C) COMMAND SEPARATION D) VERIFY CAPSULE PROGRAM E) VERIFY SEPARATION F) TRACK CAPSULE & SPACECRAFT	A) TRANSMIT ORB • ROLL MAGNI • YAW MAGNI • MOTOR BUR B) TRACK FLIGHT C) RECEIVE, REDU D) PROVIDE COM
ECEIVE ROLL & YAW TURN COMMAND & RELAY TO CC&S ECEIVE ANTENNA POSITIONING DATA & RELAY TO CC&S ECEIVE MOTOR BURN DURATION AND & ΔV RELAY TO CC&S ECEIVE INITIATE MANEUVER COMMAND & RELAY TO CC&S WITCH TO DATA MODE NO. 1 (FOR LATE MIDCOURSE MANEUVERS) ELAY VERIFICATION OF CORRECTION MANEUVER TO DSN TORE AND/OR TRANSMIT ENGINEERING DATA TO DSN (MODE 3) RANSMIT ENGINEERING (SPACECRAFT & CAPSULE) + CRUISE SCIENCE O DSN (MODE 2)	A) RECEIVE PRESEPARATION SEQUENCE COMMANDS & RELAY TO CC&S B) SWITCH TO DATA MODE NO. 2 C) TRANSMIT CAPSULE ENGRG DATA TO DSN D) RECEIVE SEPARATION COMMAND FROM DSN & ROUTE TO CC&S E) STORE AND/OR TRANSMIT ENGINEERING DATA TO DSN (MODE 3) TRANSMIT ENGINEERING (SPACECRAFT & CAPSULE) + CRUISE SCIENCE TO DSN (MODE 2)	A) RECEIVE ROLL TO CC&S B) RECEIVE ANTEN C) RECEIVE MOTO D) RECEIVE CAPSU E) TRANSMIT FLIG SCIENCE DATA
OMMAND TELECOMMUNICATION TO DATA MODE NO. 1 ECEIVE TRAJECTORY CORRECTION PARAMETERS WITCH GYROS TO RATE CONTROL OMMAND ANGULAR MANEUVER ECEIVE GYRO SIGNALS & SUM FOR ANGULAR MANEUVERS WITCH ATTITUDE REFERENCE TO AUTOPILOT SUBSYSTEM RM PROPULSION SYSTEM ROVIDE COMMAND TO INITIATE ATTITUDE MANEUVER & ΔV MANEUVER ECEIVE ACCELEROMETER SIGNALS & INTEGRATE FOR ΔV MANEUVER ISARM PROPULSION SYSTEM OMMAND REVERSE ANGULAR MANEUVER OMMAND REACQUIRE CELESTIAL REFERENCES OMMAND AUTOPILOT SELECTION CONTROL ACKUP COMMANDS AS REQUIRED	A) LOAD CAPSULE WITH 5 OR MORE COMMANDS B) COMMAND CAPSULE SEQUENCING C) COMMAND TELECOMM SWITCH TO DATA MODE NO. 1 D) COMMAND GYROS ACTIVATE (CAPSULE) E) COMMAND GYROS ESTABLISH POLARITY (CAPSULE) F) COMMAND GYROS TO ATTITUDE CONTROL MODE (CAPSULE) G) COMMAND DATA TAPE RECORDER OFF H) COMMAND CAPSULE ACTIVATE ENTRY SCIENCE I) COMMAND CAPSULE ACTIVATE TELE-COMM & VERIFY RADIO LINK (VHF) J) SELECT T/M MODE FOR TRANSMISSION OF CAPSULE DATA K) COMMAND: BIOBARRIER SEPARATION S/C ORIENT TO SEPARATION ATTITUDE ELECTRICAL CONNECTION SEPARATION VERIFY SEPARATION L) COMMAND REORIENT S/C & REACQUIRE CELESTIAL REFERENCES M) BACKUP COMMANDS AS REQUIRED N) RECEIVE S/C ORIENTATION PARAMETERS	A) COMMAND SW B) COMMAND CA C) COMMAND AN D) RECEIVE ORBI E) PROVIDE SIG (CRUISE SCIE F) SWITCH ON D RECEIVE CAPS G) BACKUP COM
WITCH TO GYRO CONTROL OLL S/C AND VERIFY AW S/C AND VERIFY UTOPILOT — PROVIDE COMMAND O TVC DURING MOTOR BURN URN ON ACCELEROMETERS ROVIDE ACCELERATION DATA O CC&S URN OFF ACCELEROMETER H) YAW BACK TO ORIGINAL ATTITUDE & VERIFY I) REACQUIRE SUN (FINE SUN SENSOR) & VERIFY J) ROLL TO ORIGINAL ATTITUDE K) LOCK ON CANOPUS & VERIFY L) SWITCH AUTOPILOT FROM GYRO CONTROL TO CELESTIAL REF CONTROL	A) SWITCH TO GYRO CONTROL B) PERFORM ROLL TURN & VERIFY C) PERFORM YAW TURN & VERIFY D) MAINTAIN ATTITUDE DURING SEPARATION E) PERFORM YAW TURN (INVERSE TO C) & VERIFY F) PERFORM ROLL TURN (INVERSE TO B) & VERIFY G) REACQUIRE CELESTIAL REFERENCES H) RETURN TO CELESTIAL REFERENCE ATTITUDE CONTROL MODE	A) UPDATE CANO B) SWITCH AUTO C) MAINTAIN S/C
OURSE PROPULSION SYSTEM RM PRESSURIZATION SYSTEM (FIRE SQUIB VALVE) RM PROPELLANT FEED SUBSYSTEM (FIRE SQUIB VALVE) IRE PROGRAMMED ENGINES (OPERATE SOLENOID VALVE) HUT DOWN ENGINE SOLATE PROPELLANT FEED SUBSYSTEM SOLATE PRESSURIZATION SUBSYSTEM		
ROVIDE CONDITIONED SOLAR OR BATTERY POWER TO: • TELEMETRY • CC&S • AUTOPILOT • ATTITUDE REFERENCE • REACTION CONTROL • PROPULSION • TEMPERATURE CONTROL • PYROTECHNIC • CAPSULE	A) PROVIDE CONDITIONED SOLAROR BATTERY POWER TO: • T/M • CC&S • AUTOPILOT • ATTITUDE REF • REACTION CONTROL • TEMPERATURE CONTROL • PYROTECHNICS • CAPSULE	A) PROVIDE CON • T/M • CC&S • AUTOPILOT • ATTITUDE R • REACTION C • TEMPERATU B) CHARGE BATTE
UPPORT S/C ASSEMBLIES UPPORT S/C COMPONENTS MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS ROVIDE ACCEPTABLE STATIC & DYNAMIC LOAD ENVIRONMENTS UPPORT FLIGHT CAPSULE	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D) PROVIDE ACCEPTABLE STATIC & DYNAMIC LOAD ENVIRONMENTS E) SUPPORT FLIGHT CAPSULE	A) SUPPORT S/C B) SUPPORT S/C C) MAINTAIN AD D) PROVIDE ACCE E) DRIVE SCAN P
IDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERA
EMOVE PROPULSION SYSTEM INHIBIT ROVIDE IGNITION NHIBIT PROPULSION SYSTEM	A) RECEIVE SIGNAL TO REMOVE CAPSULE SEPARATION INHIBIT B) RECEIVE SIGNAL TO FIRE CAPSULE UMBILICAL SEPARATION C) RECEIVE SIGNAL TO FIRE BIOBARRIER REMOVAL D) RECEIVE SIGNAL TO FIRE CAPSULE SEPARATION E) RECEIVE SIGNAL FOR SEPARATION INHIBIT	RECEIVE SQUIB F OF SCAN PLATFOR
ECEIVE CC&S COMMAND "CRUISE SCIENCE OFF" URN OFF CRUISE SCIENCE	A) RECEIVE CC&S COMMAND "CRUISE SCIENCE OFF" B) TURN OFF CRUISE SCIENCE	A) RECEIVE CRUI B) ACQUIRE CRU C) TRANSFER CO D) RECEIVE POWE E) RECALIBRATE
		A) RECEIVE DAE B) PROCESS & TR

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APSULE
DRY
ON → 9.0 FLIGHT CAPSULE CRUISE → 10.0 FLIGHT CAPSULE ENTRY → 11.0 FLIGHT CAPSULE DESCENT & LANDING → 12.0 LA
OPERATI

8.0 FLIGHT S/C CRUISE → 13.0 FLIGHT S/C ORBIT INSERTION → 14.0
OPE

INJECTION PARAMETERS IDE & DIRECTION IDE & DIRECTION START TIME PACECRAFT E & DISPLAY SCIENCE & ENGINEERING DATA ANDS TO BACK UP CC&S AS REQUIRED	A) RECEIVE, DECODE, & DISPLAY ENGINEERING DATA B) RELAY UPDATED ORBIT INJECTION PARAMETERS C) PROVIDE COMMAND TO INITIATE INSERTION MANEUVER D) PROVIDE BACKUP COMMANDS TO CC&S AS REQUIRED E) TRACK S/C F) VERIFY ORBIT INSERTION	A) RECEIVE, DECODE, & DISPLAY S B) RECEIVE AND ANALYZE ENGINEE C) TRACK S/C D) RECEIVE & DISPLAY CAPSULE D E) PROVIDE DATA FOR ORBITAL TR F) RECEIVE VERIFICATION OF ORB G) TERMINATE MISSION
YAW TURN MAGNITUDE & DIRECTION COMMAND & RELAY IA POSITIONING DATA & RELAY TO CC&S -BURN START TIME, RELAY TO CC&S E DATA, CONDITION & RELAY TO GROUND IT SPACECRAFT ENGINEERING, CAPSULE ENGRG, & O GROUND VIA MODE 2	A) UPDATE ROLL-TURN MAGNITUDE & DIRECTION COMMAND & RELAY TO CC&S B) UPDATE YAW-TURN MAGNITUDE & DIRECTION COMMAND & RELAY TO CC&S C) UPDATE MOTOR-BURN START TIME & RELAY TO CC&S D) UPDATE INITIATE MANEUVER & RELAY TO CC&S E) SWITCH TO DATA MODE NO. 2 OR 5A AS REQUIRED F) REACQUIRE EARTH AFTER ROLL TURNS & MOTOR BURN G) RELAY VERIFICATION OF MANEUVERS	A) TRANSMIT SCIENCE & ENGINEER B) RECORD ENGINEERING & SCIEN C) RECEIVE, STORE & RELAY CAPS D) RECEIVE & RELAY DATA FOR ORI
ICH TO DATA MODE NO. 2 IPUS CONE ANGLE SETTING NNA STEP (AS REQUIRED) INJECTION PARAMETERS AL TO PYRO & MECHANISMS TO DEPLOY SCAN PLATFORM E T/M OFF) & SWITCH TO DATA MODE NO. 1 A RECORDER-COMMAND TELECOMMUNICATIONS LE DATA ANDS AS REQUIRED	A) UPDATE STORED ROLL, PITCH, & YAW MAGNITUDE & SIGN B) UPDATE STORED VELOCITY MAGNITUDE C) INITIATE MANEUVER SEQUENCE D) COMMAND ORIENTATION OF S/C TO INJECTION ATTITUDE E) COMMAND THRUST FOR ORBIT INSERTION F) COMMAND RETURN TO CRUISE ATTITUDE G) BACKUP COMMANDS AS REQUIRED	A) COMMAND ORBITAL TRIM MAN B) SWITCH DATA MODES AS NECES C) RECEIVE ANY ORBITAL OPERAT D) COMMAND POSITIONING OF SC E) SWITCH ON ORBITAL SCIENCE F) SELECT RECORDED SCIENCE RE G) BACKUP COMMANDS AS REQUI H) STORE DATA FOR ORBIT TRIM (I) TERMINATE MISSION ON COMM
US CONE ANGLE ON COMMAND ILOT TO CRUISE MODE ATTITUDE TO CELESTIAL REFERENCES DURING CRUISE	A) SWITCH TO GYRO CONTROL B) PROVIDE ROLL TO PROPER ATTITUDE & VERIFY ROLL C) YAW TO PROPER ATTITUDE & VERIFY D) AUTOPILOT – PROVIDE COMMAND TO TVC DURING MOTOR BURN E) TURN ON ACCELEROMETER F) PROVIDE ACCELEROMETER DATA TO CC&S G) TURN OFF ACCELEROMETER H) YAW BACK AFTER MOTOR BURN I) ACQUIRE SUN & VERIFY J) ROLL BACK TO PROGRAMMED ATTITUDE K) ACQUIRE CANOPUS & VERIFY L) SWITCH AUTOPILOT FROM GYRO – CONTROL TO CELESTIAL REFERENCE CONTROL	A) MAINTAIN S/C ATTITUDE TO CE B) SWITCH AUTOPILOT TO GYRO C C) REACQUIRE CELESTIAL REFEREN D) REORIENT S/C FOR ORBITAL TR E) PROVIDE TVC DURING ENGINE F) REACQUIRE CELESTIAL REFEREN G) UPDATE CANOPUS CONE ANGLE
	ORBIT INSERTION ENGINE A) ARM TVC B) ARM IGNITER C) FIRE MOTOR & PROVIDE THRUST & TVC D) TERMINATE THRUST (MOTOR BURNS TO DEPLETION)	MIDCOURSE A) ARM PRESSURIZATION & PRO B) PROVIDE THRUST (FIRE PROGR IF REQUIRED C) TERMINATE THRUST ON COMMA D) ISOLATE PROPELLANT FEED SU E) ISOLATE PRESSURIZATION SU
ITIONED SOLAR ELECTRICAL POWER TO: F NTROL E CONTROL IES	A) PROVIDE CONDITIONED SOLAR OR BATTERY POWER TO: • TELEMETRY • CC&S • ATTITUDE REFERENCE SUBSYSTEM • AUTOPILOT • TEMPERATURE CONTROL • PULSE TO PYROTECHNICS • PULSE TO PROPULSION	A) PROVIDE CONDITIONED SOLAR • TELEMETRY • CC&S • ATTITUDE REFERENCE SUBSY • AUTOPILOT • TEMPERATURE CONTROL • STRUCTURES & MECHANISM • SCIENCE
SSEMBLIES OMPONENTS QUATE ALIGNMENT BETWEEN COMPONENTS TABLE STATIC & DYNAMIC LOAD ENVIRONMENTS ATFORM TO DEPLOYED POSITION & LOCK	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D) PROVIDE ACCEPTABLE STATIC & DYNAMIC LOAD ENVIRONMENTS	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNME D) PROVIDE ACCEPTABLE STATIC ENVIRONMENTS E) POSITION SCAN PLATFORM
URE CONTROL	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL
RING SIGNAL FOR UNLATCHING A	A) RECEIVE & EXECUTE COMMAND SIGNAL TO REMOVE INHIBIT FOR PROPULSION ENGINE B) RECEIVE & EXECUTE SIGNAL FOR PROPULSION ENGINE START C) RECEIVE & EXECUTE COMMAND SIGNAL FOR PROPULSION ENGINE STOP D) RECEIVE & EXECUTE SIGNAL FOR PROPULSION INHIBIT	
E SCIENCE "ON" COMMAND E SCIENCE DATA ITIONED DATA TO DAE FROM E/P CIENCE EXPERIMENTS AS REQUIRED	SCIENCE OFF DURING ORBIT INSERTION	A) TURN ON ORBITAL SCIENCE I B) ACQUIRE ORBITAL SCIENCE D C) TRANSFER CONDITIONED DAT D) RECALIBRATE SCIENCE EXPE
N" COMMAND NSFER DATA TO TM		A) RECEIVE DAE "ON" COMMAND B) PROCESS & TRANSFER DATA

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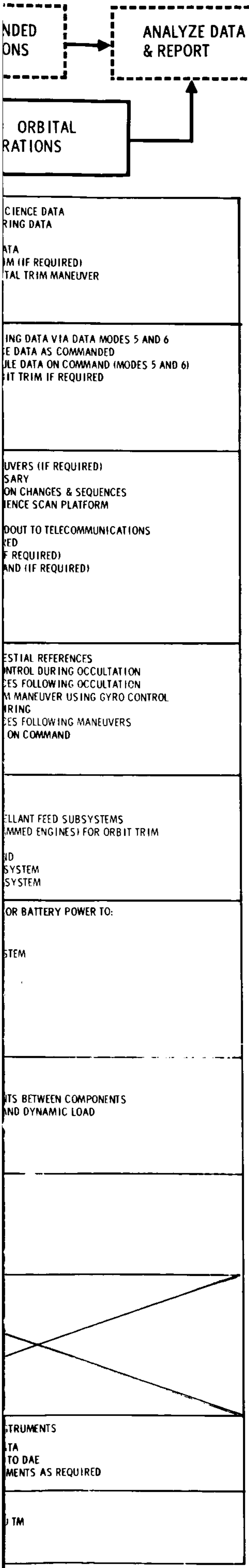


Figure 3.9-2 Flight Sequence Subsystem Correlation Chart

5

events within the described flight sequence can be altered by reprogramming the CC&S. Reprogramming of the CC&S can be accomplished at any time prior to launch or while the spacecraft is in flight. Specifically attitude maneuver sequences may be either fully automatic or ground sequenced or any combination between in accordance with the desired mission profile. For example a fully automatic midcourse maneuver sequence is shown in Paragraph 3.9.3. This sequence can be programmed in place of those described in Paragraph 3.9.2. The automated approach would greatly reduce the ground to air transmission and time-off-sun during maneuver sequences.

Launch and Injection--Launchings for the 1971 Planetary Vehicle will take place after AFETR using the Saturn S-IB/Centaur 3-stage launch vehicle. From liftoff until shroud ejection communication from the Flight Spacecraft will be through a parasitic antenna located in the shroud. After shroud ejection, communications will be maintained via the Flight Spacecraft low-gain antenna.

A parking orbit ascent mode shall be utilized for the Mars 1971 mission. An arbitrary limit of 25-minute parking orbit presently exists for the 1971 mission; the minimum parking orbit coast time will be two minutes.

The launch vehicle will inject the Planetary Vehicle on a trans-Mars trajectory and will provide the signal to initiate separation of the spacecraft from the Centaur stage. Dispersions of the Planetary Vehicle, produced by the launch vehicle, are to be correctable by a maximum 1-sigma midcourse velocity increment of 15 meters/second applied two to ten days after injection.

Initial Acquisition--After separation of the Planetary Vehicle from the Centaur stage, the Centaur stage will decelerate by employing retro-thrust. Immediately after separation, solar panels, high and low-gain antenna, and the magnetometer boom will be deployed and the science scan platform unlatched. The Planetary Vehicle will rotate in yaw and pitch maneuvers to acquire a Solar reference fix, and then be programmed through roll for acquisition of Canopus. Power during boost and acquisition will be supplied by batteries. Solar acquisition will nominally be completed within 20 minutes after injection.

Cruise Phase--During the cruise, the Planetary Vehicle will remain attitude stabilized. Continuous operational coverage for the Planetary Vehicle during the cruise phase will be supplied by the Deep Space Network.

The Flight Spacecraft, except during maneuvers, will accept data at the rate of 10 bits/second from the Flight Capsule and transmit it to Earth. The transmitted data will consist of commutated engineering data frames alternated with science data frames. The Flight Spacecraft can transmit at least five commands to the Flight Capsule before separation. The commands may be Flight Spacecraft-stored commands and/or ground commands transmitted via the Flight Spacecraft.

Midcourse Maneuvers--The Planetary Vehicle will have the capability to perform at least four midcourse corrections. Sufficient total midcourse velocity increment will be available to correct trajectory dispersions; the required increment is approximately 75 meters/second. The first midcourse maneuver may occur as early as two days after launch. The commands for midcourse corrections will be computed on the ground, transmitted to the spacecraft and stored in the CC&S until initiation of the maneuver. The required data is:

1) Roll turn magnitude and direction
2) Yaw turn magnitude and direction
3) Δ V
4) Thrust time for backup termination of thrust in case of malfunction of the accelerometers
5) Maneuver start time
6) Antenna Repositioning Commands (when required).

Flight Capsule--Flight Spacecraft Separation--The Flight Capsule is mounted forward of the Flight Spacecraft, and interfaces with the Flight Spacecraft at the field joint between the two units. The flight separation joint is contained within the capsule-adapter, with the flight separation forward of the field joint. Flight Capsule separation will involve separation of the bio-barrier front half, orientation of the Planetary Vehicle for capsule separation, separation of the capsule and reorientation to the curise attitude after which the aft portion of the barrier will be jettisoned.

Orbit Insertion--Orbit insertion and capsule entry, descent and landing is to occur within view of the DSS at Goldstone, California. The nominal retromaneuver sequence for orbit insertion is as follows.

1) A discrete command to execute the start of the retromaneuver is sent from Earth;
2) The spacecraft will then perform a roll, reacquisition of Earth with the high-gain antenna and a verification of the roll maneuver;

3) The spacecraft will then yaw (max 75°) and verify the yaw maneuver and proceed with retro motor burn; and

4) The yaw and roll will then be performed in inverse order by the spacecraft and reacquisition of Earth and Canopus accomplished.

The nominal orbit insertion maneuver results in an off-Sun time of 45 minutes.

Orbit Sequence--Orbital parameters will be determined within the first few orbits around Mars. Following celestial reference reacquisition, the scan platform will be positioned and the planetary science acquisition program initiated. Data acquisition, recording playback time and transmission modes will be selected and sequenced based on onboard logic and ground commands.

3.9.2.2 Detailed Sequence of Events

The accompanying flow diagrams and flight sequence charts show the flight sequence for nominal 1971 Voyager mission.

Flow Diagrams--Those phases of the mission profile which are directly applicable to the Flight Spacecraft (phases 1 through 6 and 8, 13 and 14) have been expanded as separate flows to present an overall sequence for the flight mission.

Flight Sequence Charts--Each flow diagram describing the Flight Spacecraft functions has a corresponding set of flight sequence charts. The flight sequence charts describe the sequence of events required to accomplish the phase of the mission under consideration. Except for the sequence from countdown through Centaur retrofire, a five column format is used.

Column 1, Events--Events are listed by number in order of occurence. Each event consists of one or more simultanious operations.

Column 2, Time--The time at which a given event occurs. Time is referenced from liftoff T = 0, from the start of a maneuver M = 0, from start of capsule separation S = 0 or from start or orbit insertion maneuver I = 0.

Column 3, Source--The subsystem executing the particular operation or operations. See Table 3.9-1 for abbreviations used.

Column 5, Comments

<u>Time Line Forms</u>--Time line forms are used to show the concurrent, overlap and sequential or other time relationships of major functions shown on the flow diagrams or events contained on the flight sequence charts. Time lines are included for the boost phase and for the first midcourse correction maneuver.

Table 3.9-1: LIST OF ABBREVIATIONS

Pyro	Pyrotechnics
CC&S	Central Computer & Sequencer
Telecom.	Telecommunications
SFOF	Space Flight Operations Facility
DSN	Deep Space Network
TVC	Thrust Vector Control
M/C	Midcourse Correction
S/C	Spacecraft
IRU	Inertial Reference Unit
LCE	Launch Complex Equipment
MOS	Mission Operational Systems
T/M	Telemeter
A/R	Attitude Reference
Power	Electrical Power System
Science	Flight Spacecraft Science Subsystem
Data Mode 1	Transmission of engineering data only
Data Mode 2	Transmission of both engineering and real-time science data time sharing the telemetry link
Data Mode 3	Transmission of real-time science only
Data Mode 4	Playback transmission of non-real time science and real-time Mode 2 data time sharing the telemetry link
R/C	Reaction Control
DSIF	Deep Space Instrument Facility
L/V	Launch Vehicle
LOX	Liquid Oxygen
Mech.	Mechanism

Table 3.9-1: LIST OF ABBREVIATIONS

TWTA	Traveling Wave Tube Amplifier
DAS	Data Acquisition System
Engr.	Engineering
DHS	Data Handling System
ΔV	Velocity Increment
DSS	Deep Space Station

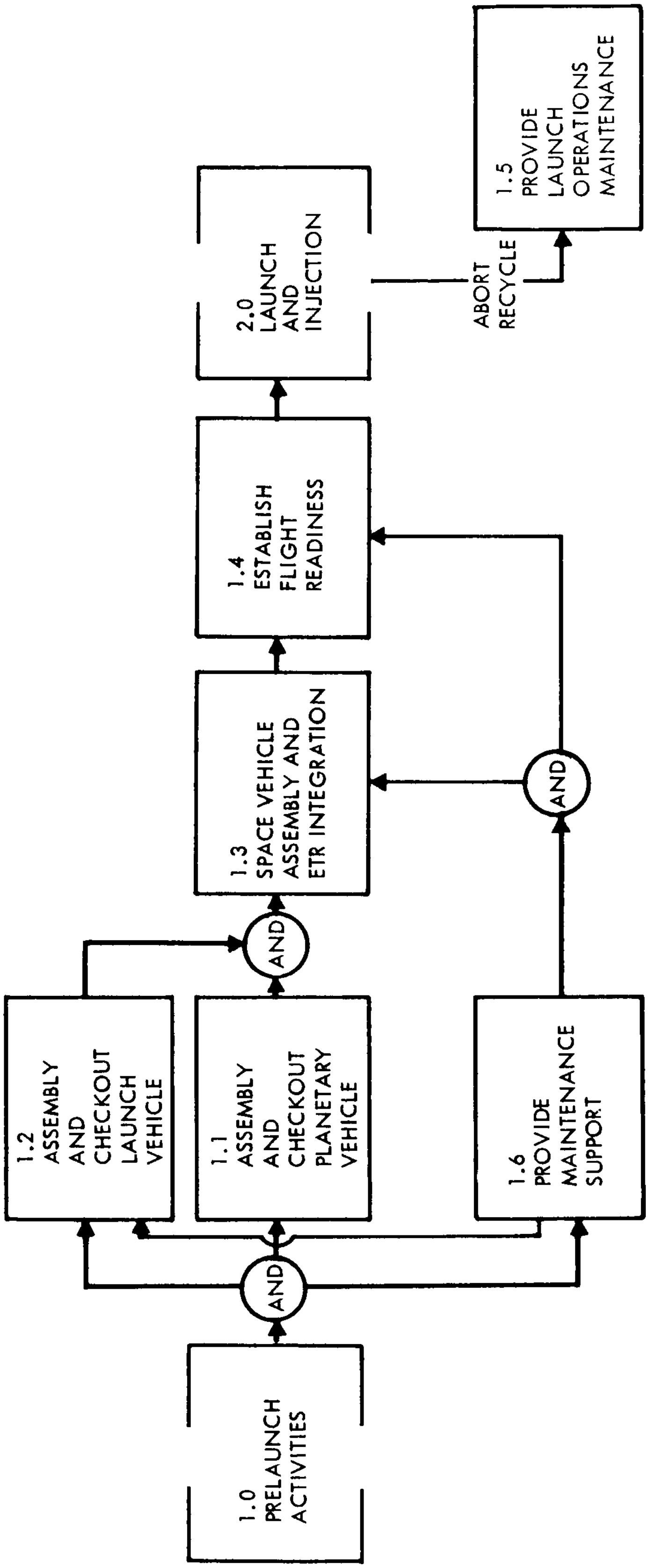


Figure 3.9-3: 1971 Mission — 1.0 Prelaunch Activities

FLIGHT SEQUENCE

EVENT	TIME*	
2.0 COUNTDOWN		
The following is a summary countdown for the Saturn S-IB/Centaur 1971 flight mission.		
This section covers the important time sequences from the initiation of COUNTDOWN (2.1) up to the start of Saturn S-IB BURN (2.16).		
1. Verify S/C air conditioning ready for power on.	T-1125*	
2. Turn on TM. Evaluate TM at DSIF 71.	T-1110	
3. Start S/C system checks, start science subsystems autocalibration, start flight capsule checks.	T-1095	
4. Turn off S/C systems.	T-885	
5. Report on S/C systems, remove rf couplers.	T-800	
6. Hold until F-0 day.	T-705	
7. Verify S/C air conditioning ready for power on.	T-670	

*All times in minutes unless noted otherwise.

FLIGHT SEQUENCE

EVENT	TIME*	
8. Turn on TM. Evaluate TM rf at DISF 71; Launch Vehicle preps.	T-660	
9. Start system checks, start science subsystems autocalibration; start flight capsule checks.	T-645	
10. Install L/V battery.	T-570	
11. S-IVB propulsion preparation.	T-550	
12. Provide external power.	T-545	
13. Verify S/C systems, switch S/C to internal power, verify S/C systems on internal power, switch S/C to external power, switch S/C power off, charge batteries	T-540	
14. Start L/V checks.	T-535	
15. Perform L/V G&C checks.	T-515	
16. Perform L/V power transfer	T-480	
17. L/V on internal power.	T-475	
18. Maintain rf silence, S-IVB/Centaur retrorocket installation, S-IB hypergolic installation.	T-470	
19. Clear pad.	T-420	

*All times in minutes unless noted otherwise.

FLIGHT SEQUENCE

EVENT	TIME*	
20. S-IB LOX loading to 18 percent	T-410	
21. S-IB mechanical preparations.	T-370	
22. Perform L/V communication checks	T-355	
23. Maintain rf silence, L/V destruct checks.	T-350	
24. S-IB initiator connection.	T-320	
25. Open service structure platform.	T-280	
26. Turn on TM. Evaluate TM at DSIF 71.	T-270	
27. Start final checks.	T-260	
28. Clear pad.	T-245	
29. Centaur LOX loaded to 100 percent.	T-240	
30. S-IVB LOX loaded to 100 percent.	T-220	
31. L/V computer on.	T-200	
32. Remove service structure	T-190	
33. Load S-IB LOX to 100 percent.	T-160	
34. S-IVB-Centaur LH2-100 percent.	T-100	

*All times in minutes unless noted otherwise.

FLIGHT SEQUENCE

EVENT		TIME*	
	35. Start terminal count.	T-35	
	36. L/V power transfer.	T-25	
	37. Switch S/C to internal power, verify systems.	T-10	
2.16	BURN S-IB STAGE		
	1. Ignite engines.	T-4	
	2. Thrust buildup	T=0	
	3. Liftoff		
	4. Boost to burnout.	T+140 sec	
	5. Terminate thrust.	T+143 sec	
2.17	SEPARATE S-IB STAGE		
2.18	BURN S-IVB STAGE	T+633 sec	
2.19	SEPARATE S-IVB STAGE		
2.20	BURN CENTAUR STAGE (FIRST BURN)	T+733 sec	
2.21	EARTH ORBIT CRUISE (2 to 25 min)	T+13 to 38 min	

*All times in minutes unless otherwise noted.

FLIGHT SEQUENCE

EVENT	TIME	
2.22 BURN CENTAUR STAGE (SECOND BURN)	T+18 to 43 min	
2.23 SEPARATE CENTAUR STAGE		
1. Separation monitoring signal to CC&S.		
2. Arm pyrotechnics		
2.24 CENTAUR STAGE RETROFIRE		
2.25 PROVIDE RANGE SAFETY		

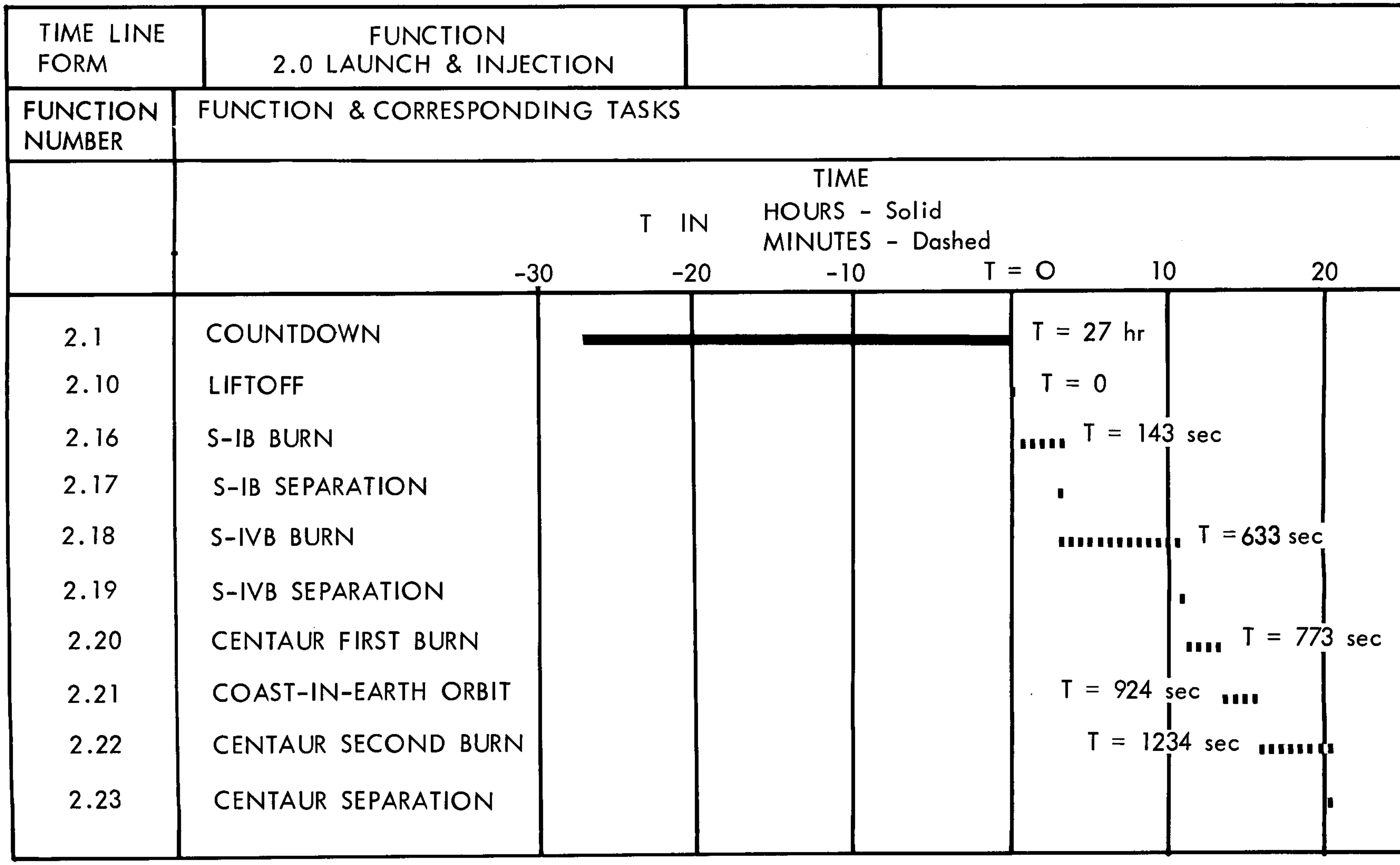
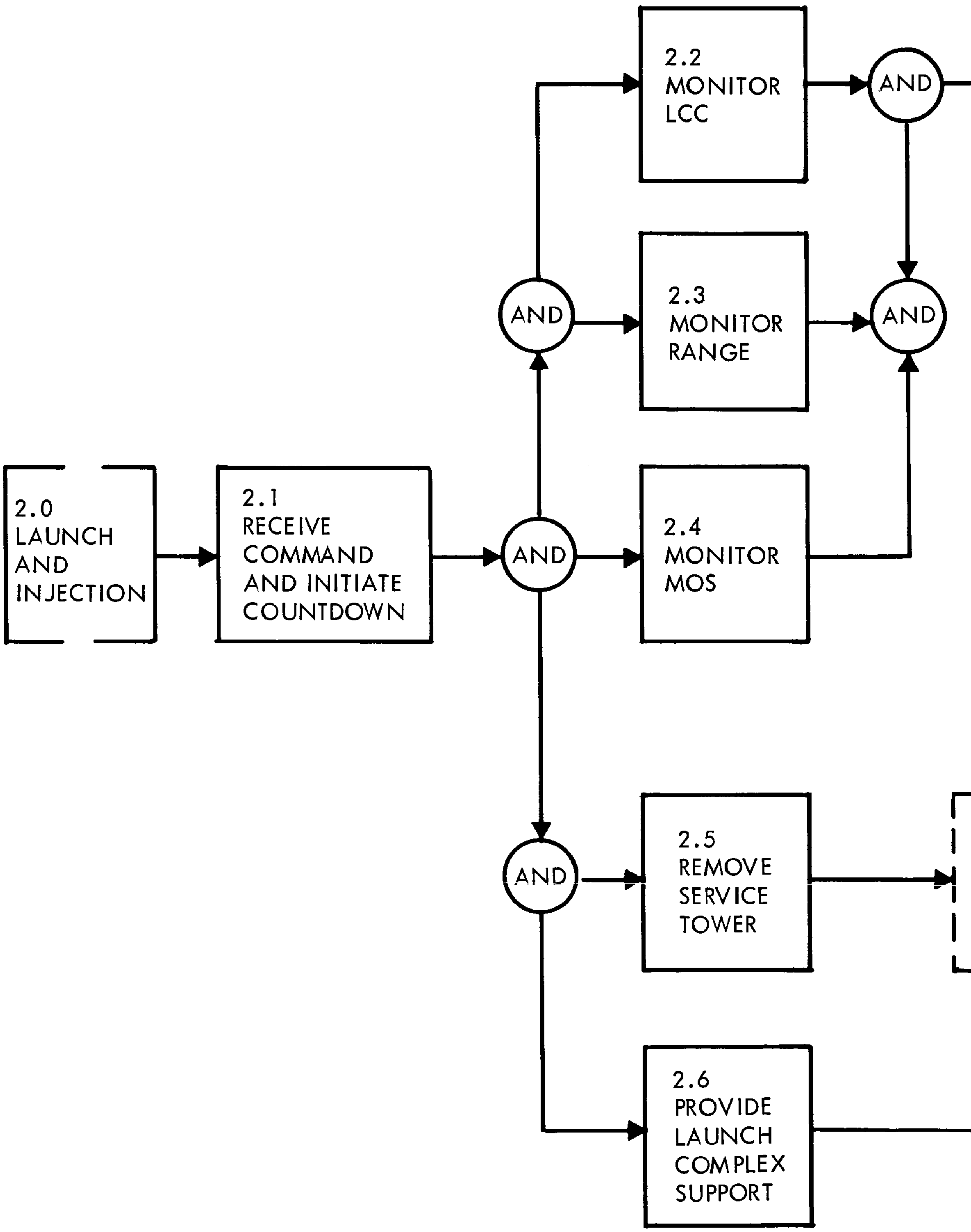
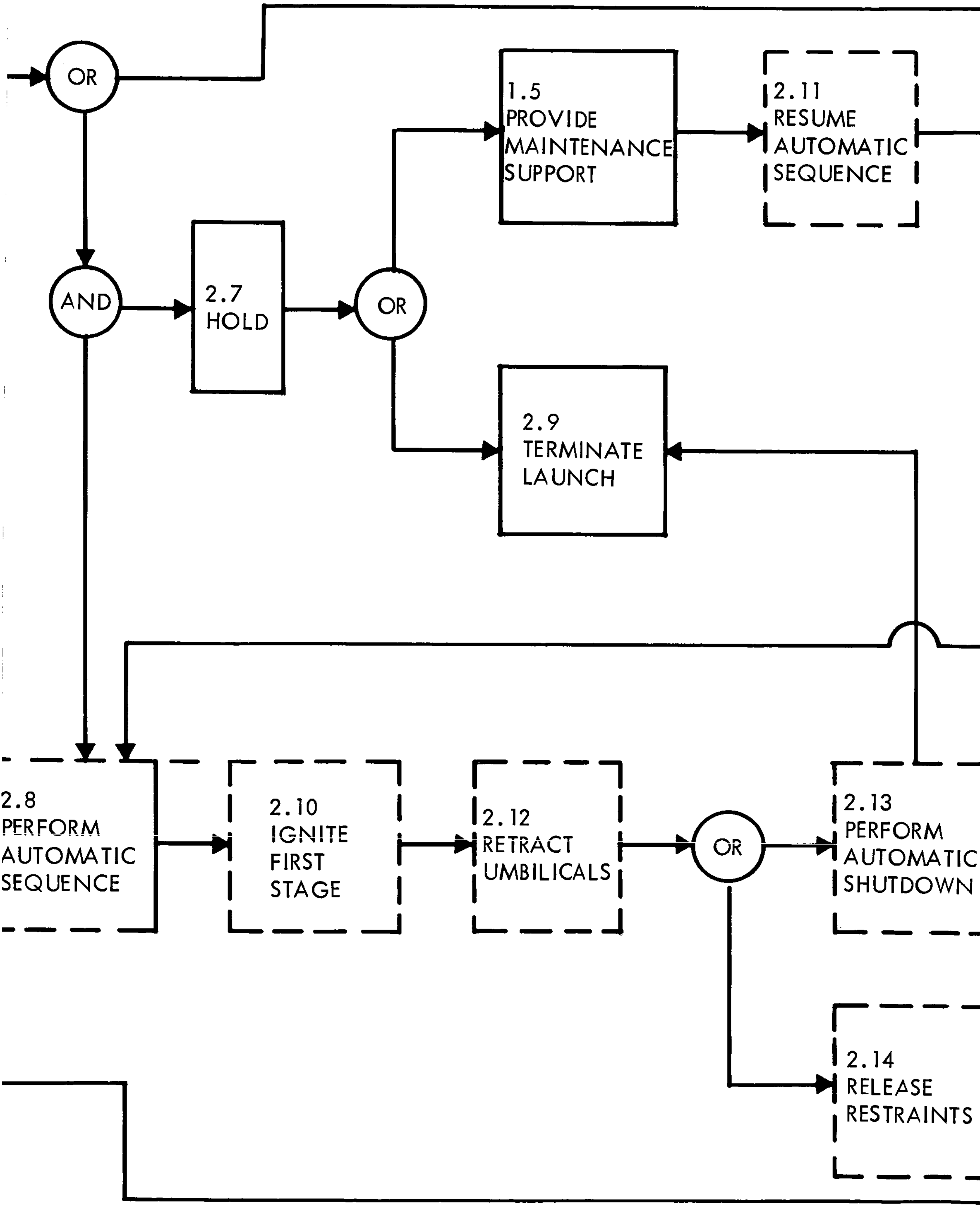
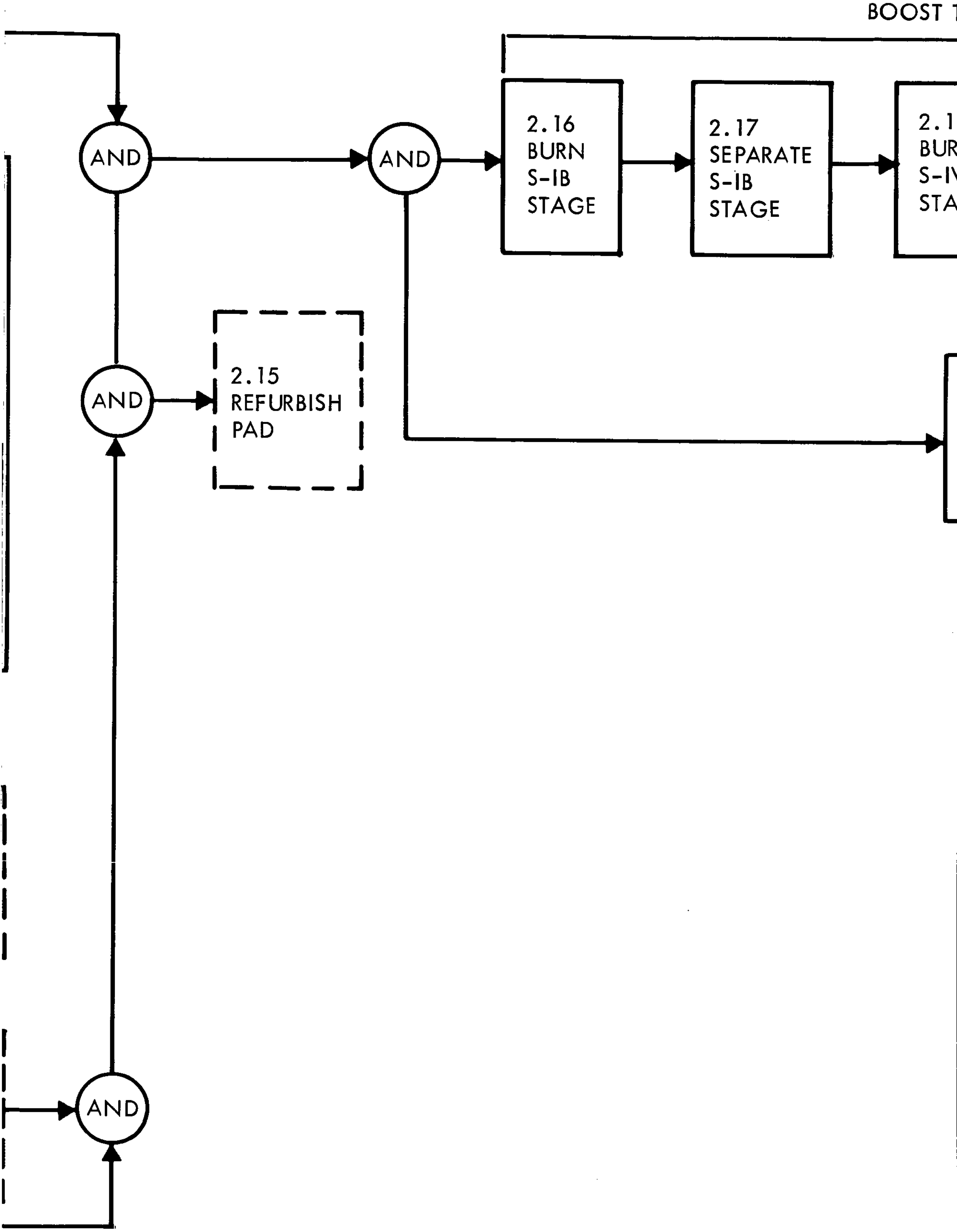


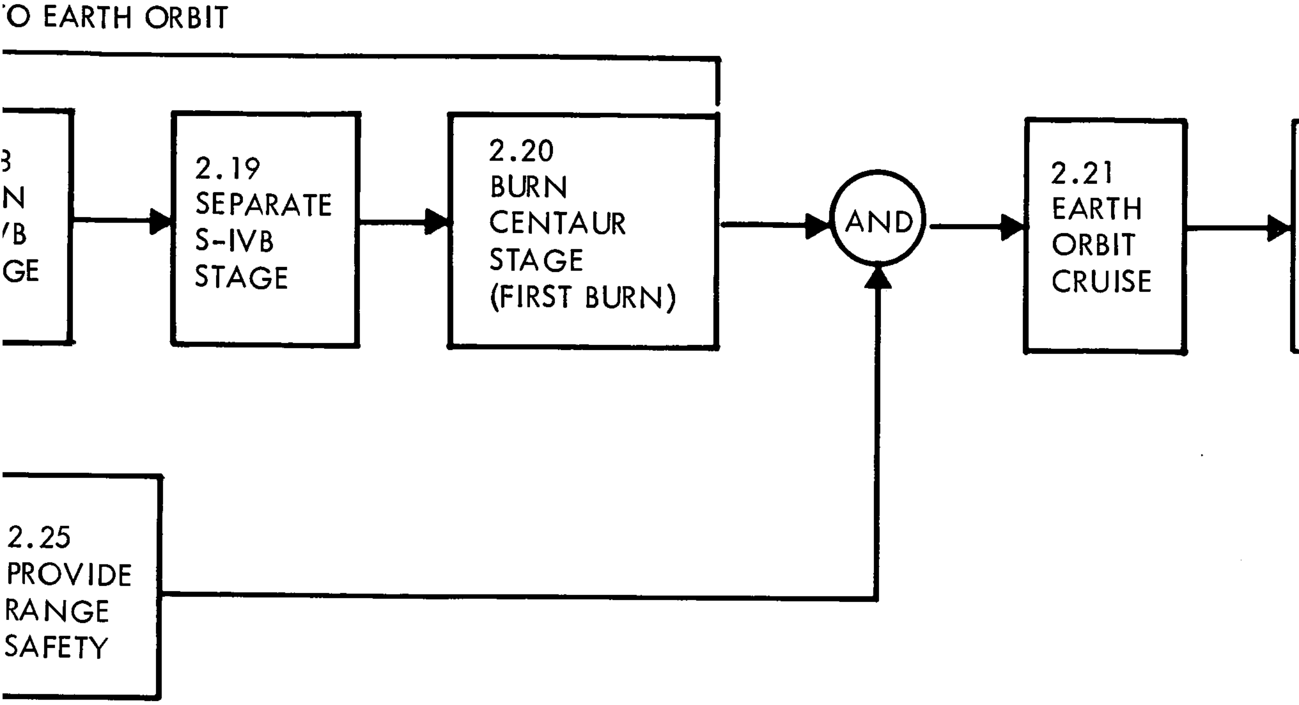
Figure 3.9-4: 1971 Mission — Time Line, 2.0 Launch & Injection


2.0
LAUNCH
AND
INJECTION
2.1
RECEIVE
COMMAND
AND INITIATE
COUNTDOWN
AND
AND
AND
AND
AND
AND
2.2
MONITOR
LCC
2.3
MONITOR
RANGE
2.4
MONITOR
MOS
2.5
REMOVE
SERVICE
TOWER
2.6
PROVIDE
LAUNCH
COMPLEX
SUPPORT


OR
AND
2.7 HOLD
OR
1.5 PROVIDE MAINTENANCE SUPPORT
2.11 RESUME AUTOMATIC SEQUENCE
2.9 TERMINATE LAUNCH
2.8 PERFORM AUTOMATIC SEQUENCE
2.10 IGNITE FIRST STAGE
2.12 RETRACT UMBILICALS
OR
2.13 PERFORM AUTOMATIC SHUTDOWN
2.14 RELEASE RESTRAINTS


BOOST T
AND
AND
2.16 BURN S-IB STAGE
2.17 SEPARATE S-IB STAGE
2.1 BUR S-I STA
AND
2.15 REFURBISH PAD
AND

175 3


'O EARTH ORBIT
3
N
'B
GE
2.19
SEPARATE
S-IVB
STAGE
2.20
BURN
CENTAUR
STAGE
(FIRST BURN)
AND
2.21
EARTH
ORBIT
CRUISE
2.25
PROVIDE
RANGE
SAFETY

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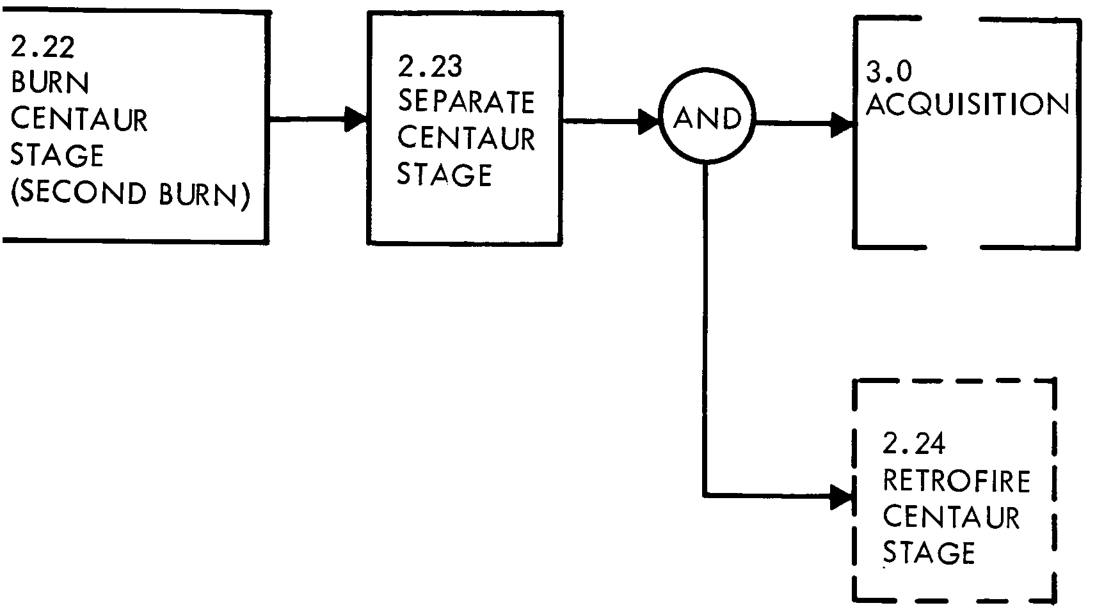


Figure 3.9-5: 1971 Mission — 2.0 Launch and Injection

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
3.0 ACQUISITION				
3.1 DEPLOY SOLAR PANELS	T+20 to 45 min			
1. Arm pyrotechnics.		CC&S	Pyro	
2. Fire solar panel squibs.		CC&S	Pyro	
3. Backup fire solar panel squibs.		CC&S	Pyro	
4. Switch solar panel motor on.		CC&S	Mechanisms	
5. Drive panels to limit stops.		Mechanisms	---	
6. Provide "out and lock" signal.		Mechanisms	Telecom	
3.2 DEPLOY ANTENNAS	T+20 to 45 min			
1. Deploy low-gain antenna.		CC&S	Pyro	CC&S stored command
a. Unlock and release		CC&S	Pyro	
b. Backup deploy signal		CC&S	Pyro	(Automatic program)
c. Move antenna to operation position and lock		CC&S	Mechanisms	
2. Verify Omniantenna deployment				
a. Provide antenna "out and lock" signal		Mechanisms	Telecom	
b. Transmit signal to space flight operations facility (SFOF)		Telecom	SFOF	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
c. Verify deployment by signal strength check		SFOF		
3. Deploy high-gain antenna				
a. Unlock and release signal		CC&S		
b. Back-up unlock and release signal				Automatic Program
c. Activate positioning mechanism		CC&S	Mechanisms	
d. Move antenna to operating position and lock		Mechanisms		2 axis servo positioning
4. Verify high-gain antenna deployment				
a. Provide antenna "out and lock" signal		Mechanisms	Telecom	
b. Transmit signal to SFOF		Telecom	SFOF	
c. Verify deployment by signal strength check		SFOF		
5. Deploy VHF antenna				
a. Unlock and release signal		CC&S	Pyro	
b. Backup release signal		CC&S	Pyro	Automatic Program
c. Move antenna to operating position and lock		Mechanisms		
3.3 DEPLOY SCIENCE ELEMENTS	T + 20 45 min			
1. Unlock and release boom signal		CC&S	Pyro	Fire Magnetometer Release Squib

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
2. Deploy science boom, magnetometer and ionization chamber		CC&S	Mechanism	Actuator
3. Activate positioning mechanism		CC&S	Mechanism	
3.4 VERIFY SCIENCE ELEMENTS DEPLOYMENT				
1. Provide "out and lock" signal		Mechanism	Telecom	
2. Transmit verification to SFOF		Telecom	SFOF	
3.5 PERFORM SOLAR ACQUISITION MANEUVERS	T + 21 to 46 min			
1. Activate reaction control latching valves squibbs		CC&S	Reaction Control	
2. Back-up activation signal, reaction control latching valves (squibbs)		CC&S	Reaction Control	Automatic Program
3. Reduce spacecraft angular rate		Autopilot	Reaction Control	Separation rates reduced
4. Enable sun sensors		CC&S	Autopilot	
5. Pitch and yaw spacecraft to sun		Attitude References (Sun Sensors)	Autopilot and Reaction Control	Maintain spacecraft alignment to sun line within $\pm$ 0.4.
(Acquisition complete)	T + 41-66 min			

FLIGHT SEQUENCE

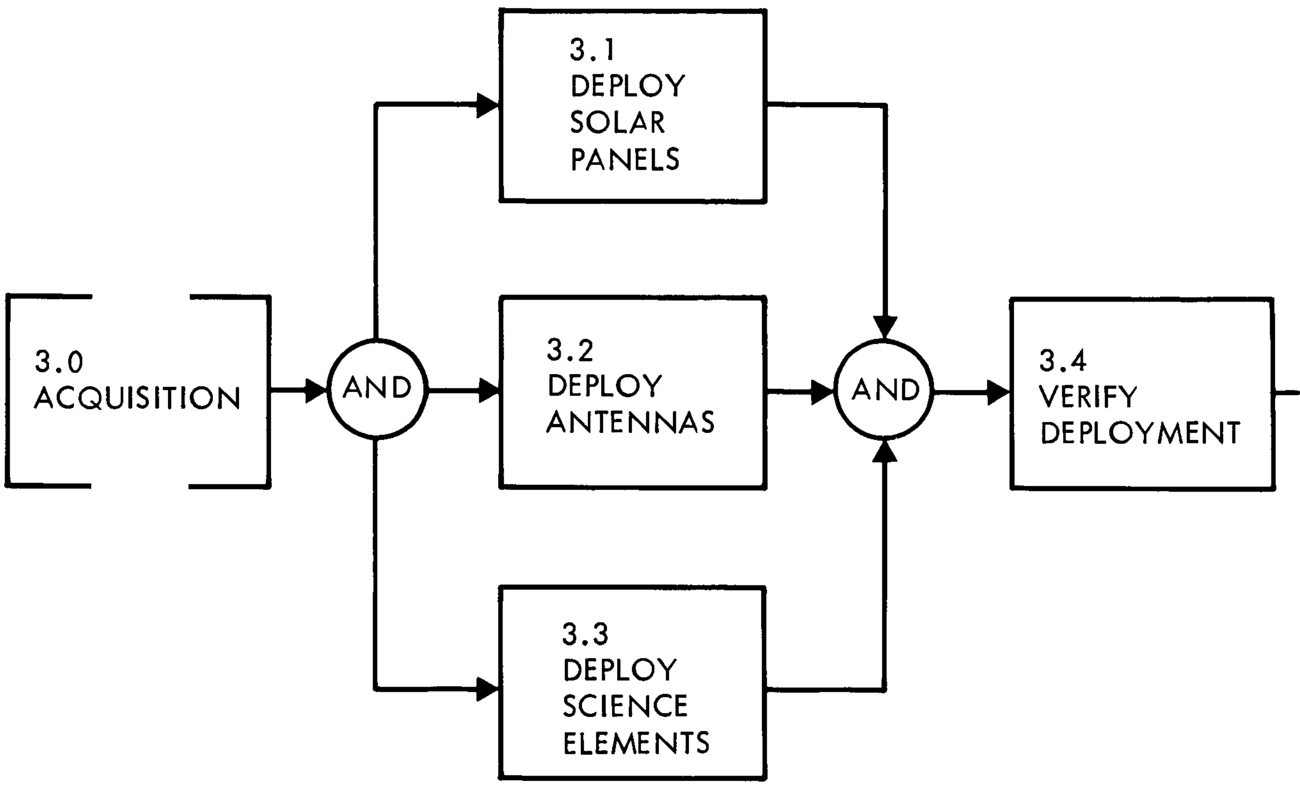
EVENT	TIME	SOURCE	DESTINATION	COMMENTS
3.6 VERIFY SOLAR ACQUISITION				
1. Sense a. Solar panel output		Electric Power	CC&S	
b. Fine sun sensor		Attitude Reference	Telecom	
2. Transmit		Telecom	SFOF	
3. Verify solar panel voltage		SFOF		
3.7 ACTIVATE SOLAR POWER SYSTEM	T + 55-80 min			
1. Switch power system to solar panel mode		Automatic	Automatic	
2. Battery charger on		Automatic	Automatic	
3.8 PERFORM CANOPUS ACQUISITION	T + 55-80 min			Canopus acquisition takes 45 min. but maneuver may start at any time up to first mic-course correction
1. Enable Canopus sensor		CC&S	Star Tracker	
2. Provide Canopus attitude search mode		CC&S	Autopilot	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
3. Activate magnetometer		CC&S	Science	
4. Command 360 spacecraft roll a. Record Canopus sensor output and prepare star map b. Calibrate magnetometer		CC&S Star Tracker Science	Autopilot	The 360 roll (30 minutes required) during Canopus acquisition will be used to calibrate the magnetometer. The output will be transmitted to DSIF. Canopus cone angle will be controlled as required. Maintain alignment to Canopus within $\pm$ 0.4. (Four cone angles used)
5. Transmit sensor output plot to SFOF.		Telecom	DSN	
6. Command spacecraft roll to align sensor with Canopus		DSN	CC&S and Autopilot	
7. Provide magnitude signal to indicate Canopus presence		Star Tracker	CC&S	
3.9 VERIFY CANOPUS ACQUISITION				
1. Transmit sensor output signal to SFOF for evaluation	T + 100 to 125 min	Telecom	SFOF	
2. Receive SFOF command to: a. To activate derived rate circuitry and deactivate gyros, or		DSN	Telecom and CC&S	Proceed to 4.0 coast

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
b. Repeat acquisition maneuver		DSN	Telecom and CC&S	Proceed to 3.10 Canopus override
3.10 PERFORM CANOPUS OVERRIDE				
1. Decode and relay override command		Telecom	CC&S	
2. Initiate roll search		CC&S	Autopilot	
3. Star acquisition signal		Star Tracker	CC&S	
4. Switch roll control to Canopus sensor error signal		CC&S	Autopilot	
5. Verify				
a. Transmit sensor output signal to SFOF		Telecom	SFOF	
b. Receive SFOF command to:				
1. Activate derived rate circuitry and deactivate gyros or		DSN	Telecom and CC&S	Proceed to 4.0 coast
2. Repeat acquisition Maneuver		DSN	Telecom and CC&S	Repeat 3.10


3.0
ACQUISITION
AND
3.1
DEPLOY
SOLAR
PANELS
3.2
DEPLOY
ANTENNAS
3.3
DEPLOY
SCIENCE
ELEMENTS
AND
3.4
VERIFY
DEPLOYMENT

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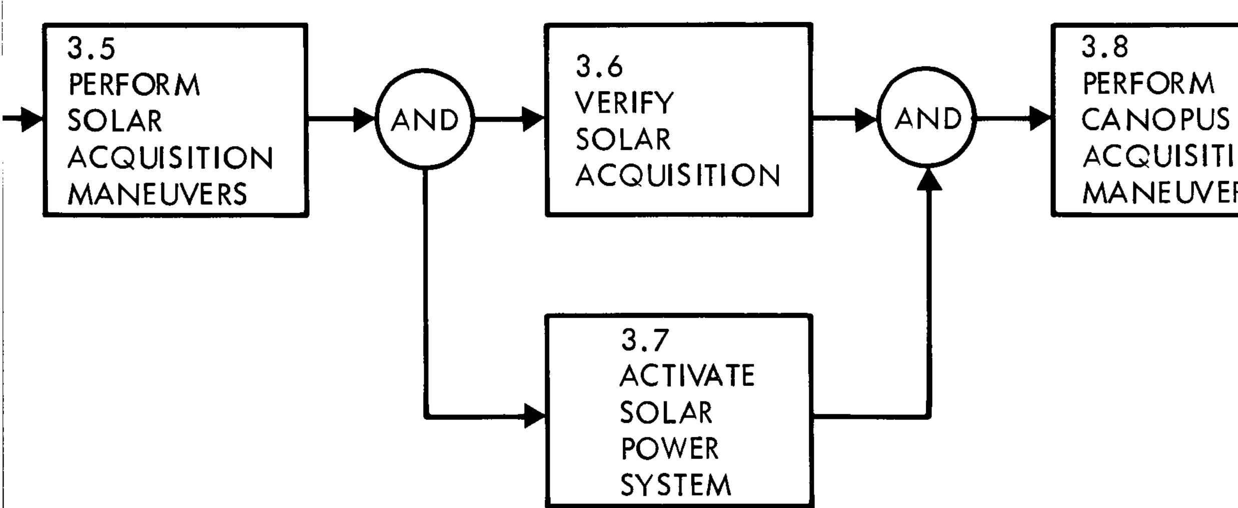
3.5 PERFORM SOLAR ACQUISITION MANEUVERS
AND
3.6 VERIFY SOLAR ACQUISITION
AND
3.8 PERFORM CANOPUS ACQUISITI
MANEUVEI
3.7 ACTIVATE SOLAR POWER SYSTEM

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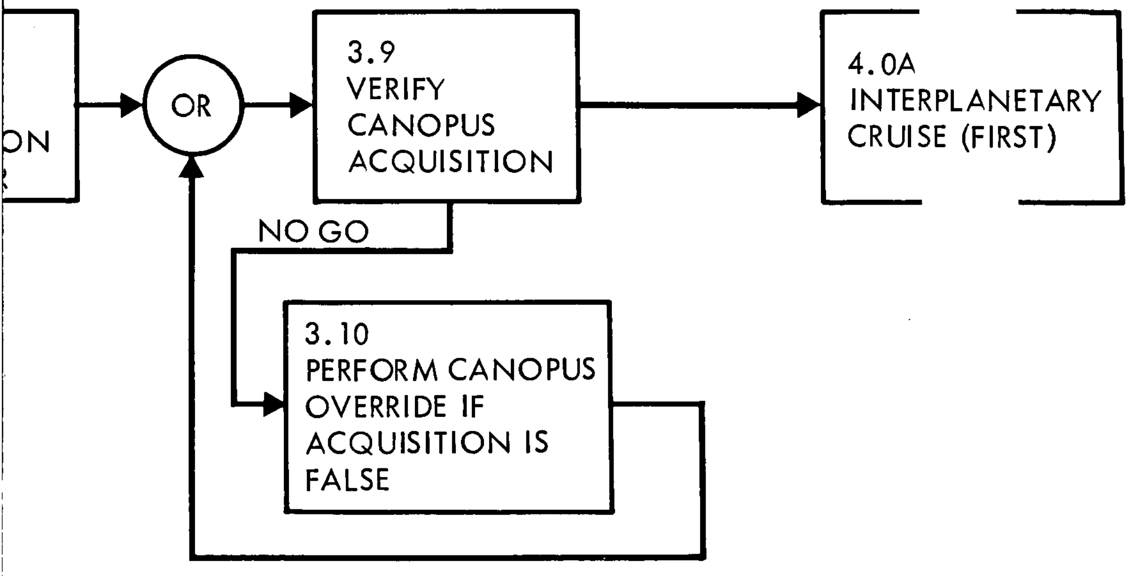


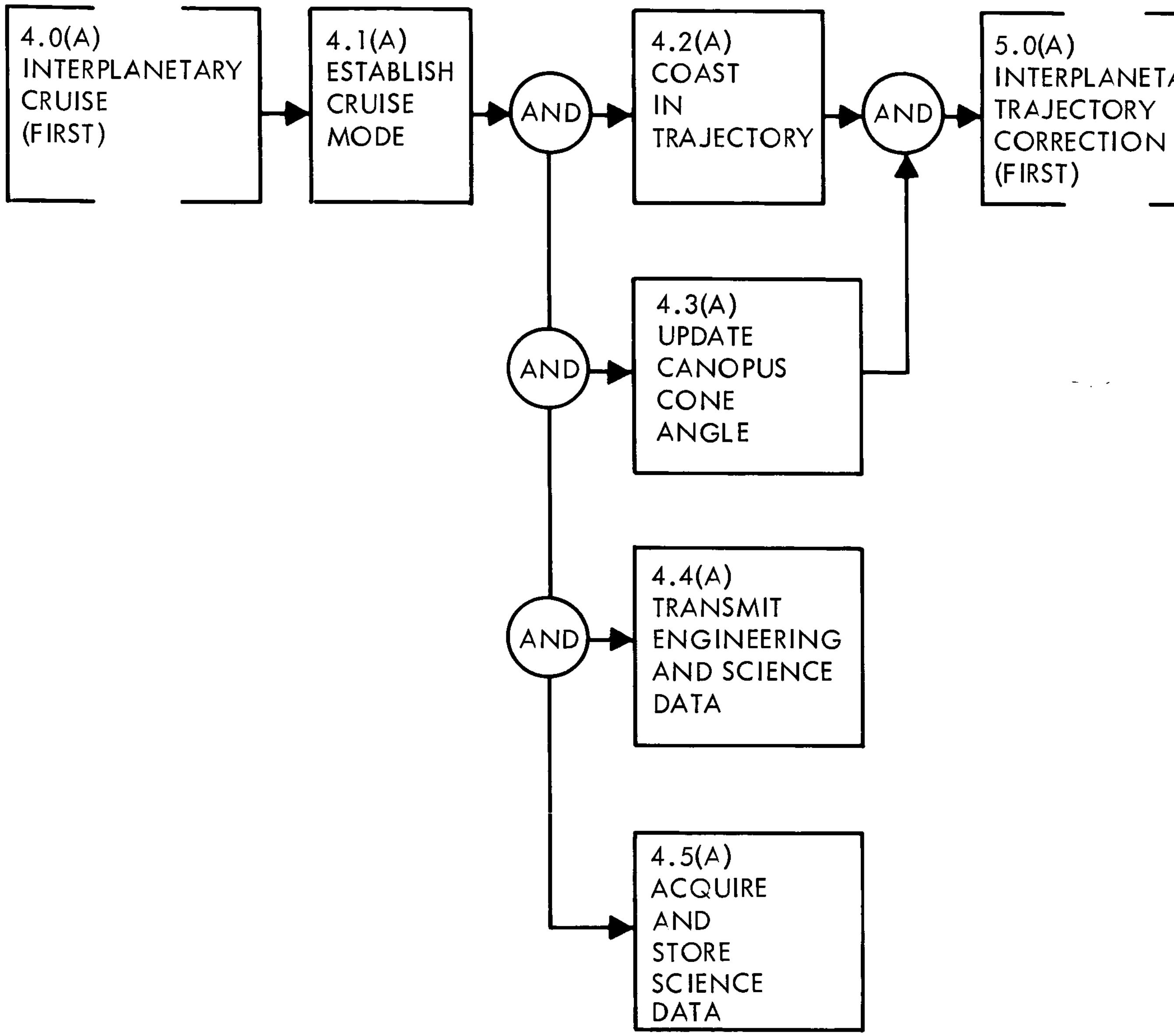
Figure 3.9-6: 1971 Mission — 3.0 Acquisition

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
4.0A FIRST INTERPLANETARY CRUISE	T + 1-10 days			
4.1A ESTABLISH CRUISE MODE				
1. Switch to exclusive use of solar panel power		Automatic	Power	
2. Switch to automatic reacuisition of celestial references (for non-catastrophic disturbances)		CC&S	Autopilot	
3. Switch to data mode 2		CC&S	Telecom	
4. Battery charger on		Automatic	Power	Cycle: 6 hours of charge required for each hour of battery discharge
5. Turn cruise science on		CC&S	Science	
6. Turn data recorder on		CC&S	Telecom	
7. Monitor capsule transit status		Capsule	Telecom	
8. Switch on TWTA	T + 1 day	CC&S	Telecom	
9. Provide power to capsule		CC&S	Power	

FLIGHT SEQUENCE

EVENT		TIME	SOURCE	DESTINATION	COMMENTS
4.2A	COAST IN TRAJECTORY				
	During first coast, the functions 4.3, 4.4 & 4.5 occur.				
4.3A	UPDATE CANOPUS CONE ANGLE				
	Set canopus cone angle No. 1	T + 9 days	CC&S	Canopus Tracker	
4.4A	TRANSMIT ENGINEERING AND SCIENCE DATA				
	Use Data Mode 2		Telecom	DSN	Data mode no. 2, 2/3 science, 1/3 engineering
4.5A	ACQUIRE AND STORE SCIENCE DATA				
	1. Initiate cruise science acquisition		CC&S	Science	
	2. Sequence science acquisition		DAS	Science	
	3. Record science data		Science	Data Recorder	
5.0A	FIRST INTERPLANETARY TRAJECTORY CORRECTION	T + 5 days (nominal)			Correction may occur anytime within 2-10 days


4.0(A) INTERPLANETARY CRUISE (FIRST)
4.1(A) ESTABLISH CRUISE MODE
AND
4.2(A) COAST IN TRAJECTORY
AND
5.0(A) INTERPLANETA TRAJECTORY CORRECTION (FIRST)
AND
4.3(A) UPDATE CANOPUS CONE ANGLE
AND
4.4(A) TRANSMIT ENGINEERING AND SCIENCE DATA
4.5(A) ACQUIRE AND STORE SCIENCE DATA

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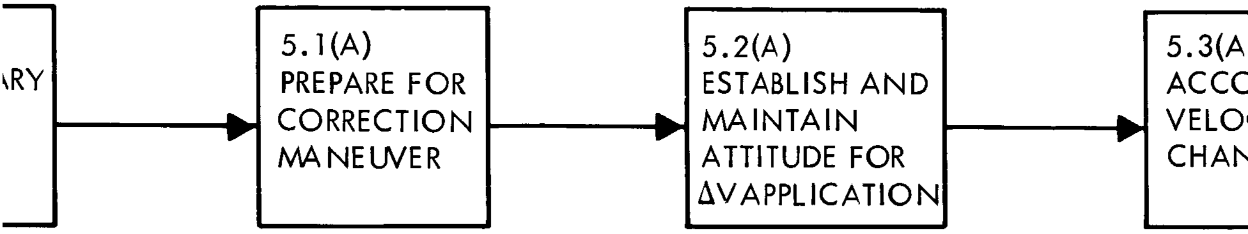
RY
5.1(A)
PREPARE FOR
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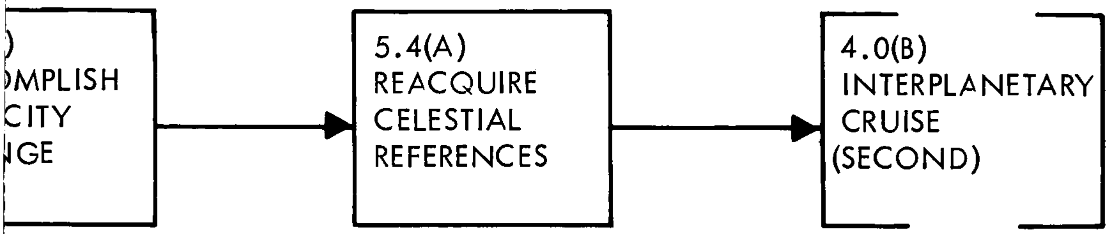
Figure 3.9-7:



1971 Mission — 4.0(A) Interplanetary Cruise (First) and
5.0 (A) Interplanetary Trajectory Correction (First)

3

TIME LINE FORM	FUNCTION 5.0 INTERPLANETARY TRAJECTORY CORRECTION (FIRST)	
FUNCTION NUMBER	FUNCTION & CORRESPONDING TASKS	
		M=0
5.1A	PREPARE FOR CORRECTION MANEUVER	TIME T NOT C
5.2A	ESTABLISH AND MAINTAIN ATTITUDE FOR ΔV APPLICATION ROLL VERIFICATION OF ROLL YAW VERIFY YAW	T=
5.3A	ACCOMPLISH VELOCITY CHANGE BURN MIDCOURSE ENGINE VERIFICATION OF BURN	
5.4A	REACQUIRE CELESTIAL REFERENCES YAW IN REVERSE (SUN ACQUIRE) VERIFY YAW ROLL (CANOPUS ACQUIRED) VERIFICATION OF CANOPUS	ON SUN
	► TOTAL AVAILABLE MIDCOURSE ENGINE BURN TIME IS 10 MINUTES	

189 (1)

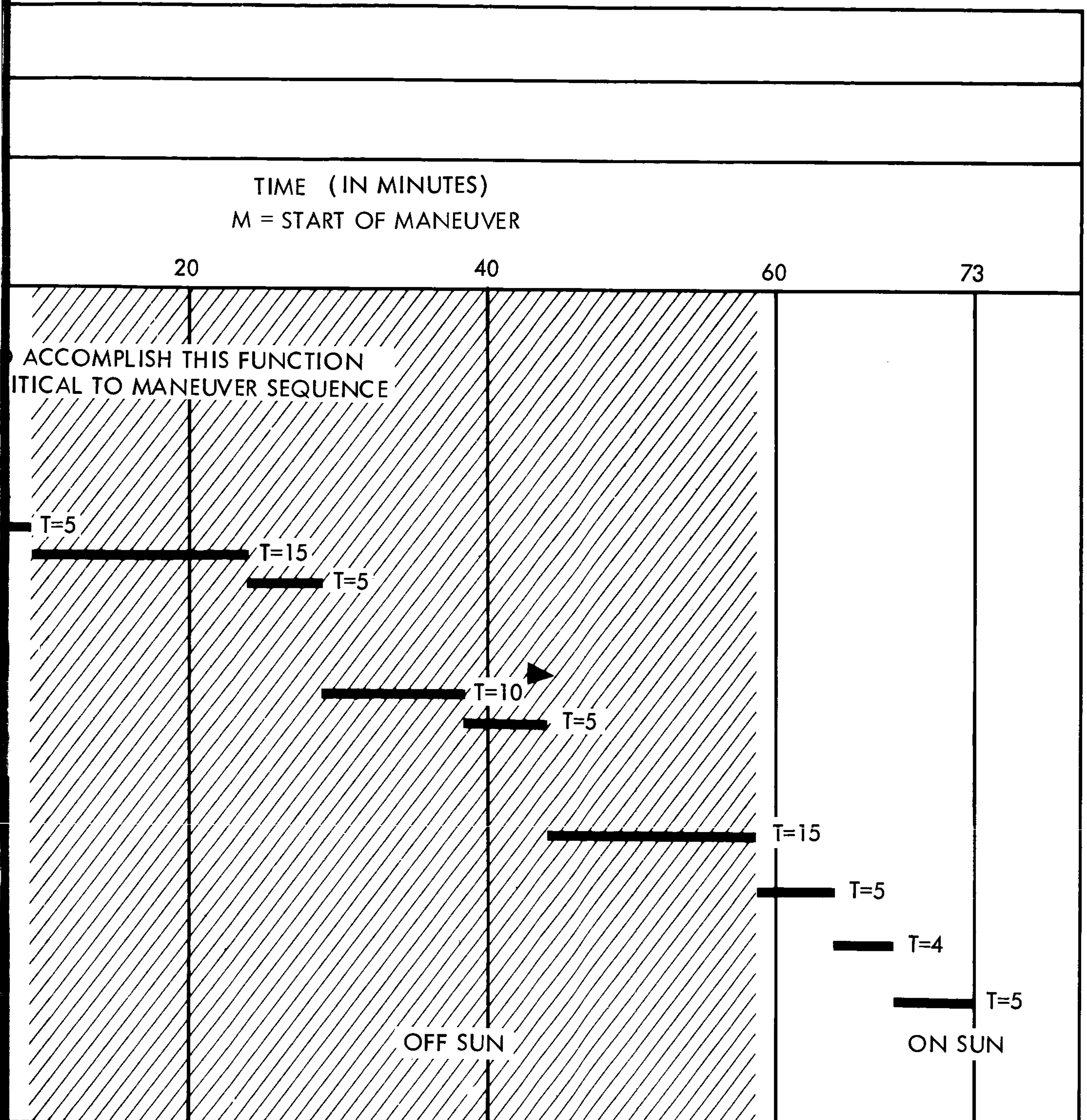


Figure 3.9-8: 5.0 Interplanetary Trajectory Correction (First)

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5.1A PREPARE FOR CORRECTION MANEUVER				
1. Receive and store:		DSN	Telecom and CC&S	
a. Roll turn magnitude and direction				
b. Yaw turn magnitude and direction				"Pitch instead of yaw is optional sequence"
c. Midcourse motor burn time				Back-up shutoff in case of accelerometer failure
d. Velocity increment				
e. Maneuver start time		DSN	Telecom and CC&S	
2. Switch to data mode #1		CC&S	Telecom	
3. Switch gyros to rate control		CC&S	Attitude Ref. via autopilot	
4. Switch autopilot to gyro reference		CC&S	Autopilot	
5. Remove propulsion inhibit and thrust vector control		CC&S	Pyro	
5.2A ESTABLISH AND MAINTAIN ATTITUDE FOR ΔV APPLICATION				
1. Decrease dead zone		CC&S	Autopilot	
2. Command roll maneuver		CC&S	Autopilot	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
3. Start roll	M = 0	Autopilot	Reaction Control	
4. Stop roll	M + 4 min	CC&S	Autopilot and Reaction Control	
5. Verification of roll				
a. Transmit degrees of roll		CC&S/ Telecom	SFOF	
b. Ground Evaluation				
c. Receive roll verified	M + 9	SFOF	Telecom/ CC&S	
6. Command yaw maneuver start	M + 9	CC&S	Autopilot	
7. Start yaw turn	M + 9	Autopilot	Reaction Control	
8. Off-sun (Switch to battery power)		Automatic		
9. Stop yaw turn	M + 24	CC&S	Autopilot and Reaction Control	
10. Verification of yaw				
a. Transmit degrees of yaw to SFOF		CC&S/ Telecom	SFOF	
b. Ground evaluation		SFOF		
c. Receive yaw verified proceed with burn	M + 29	SFOF	Telecom/ CC&S	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
11. Increase dead zone		CC&S	Autopilot	
5.3A ACCOMPLISH VELOCITY CHANGE				
1. Turn on jet vane actuators and accelerometers and select motors		CC&S	Autopilot	
2. Ignite midcourse motor	M + 29 min	CC&S	Propulsion	Maximum total time for engine burn time is 10 minutes.
3. Count accelerometer pulses and terminate midcourse engine thrust	M + 39 min	CC&S	Propulsion	
4. Terminate midcourse engine thrust and inhibit		Timed (CC&S)	Propulsion	Back-up for accelerometer malfunction
5. Verification of burn				
a. Transmit data to SFOF		CC&S Telecom	SFOF	
b. Ground evaluation		SFOF		
c. Receive burn verified, proceed with yaw	M + 44 min	SFOF	Telecom/ CC&S	
5.4A REACQUIRE CELESTIAL REFERENCES				
1. Decrease dead zone		CC&S	Autopilot	Perform attitude changes in reverse to those accomplished prior to maneuver

FLIGHT SEQUENCE

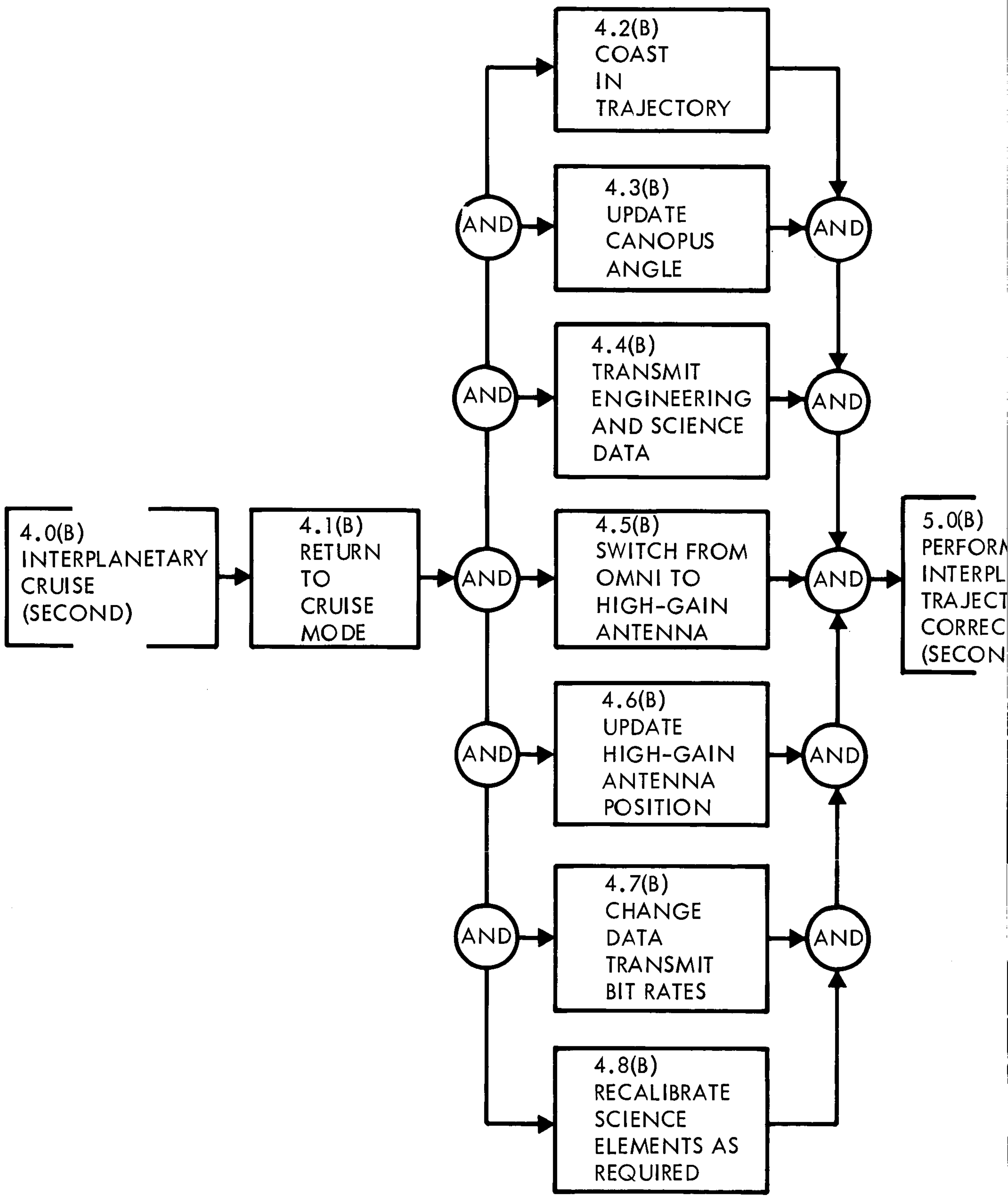
EVENT	TIME	SOURCE	DESTINATION	COMMENTS
2. Command yaw maneuver start		CC&S	Autopilot	
3. Start yaw turn	M + 44 min	Autopilot	Reaction Control	
4. Stop yaw turn	M + 59 min	CC&S	Autopilot Reaction Control	
5. Verify Sun acquisition complete				
a. Sense Sun acquisition	M + 59 min.			
1. Solar panel output		Electrical Power	CC&S	
2. Fine Sun sensor		Attitude ref	Telecom	
b. Transmit to SFOF		Telecom	SFOF	
c. Ground evaluation		SFOF		
d. Receive Sun acquisition verified, proceed with roll	M + 64 min.	SFOF	Telecom/ CC&S	
6. Command roll maneuver start	M + 64 min.	CC&S	Autopilot	
7. Start roll turn	M + 64 min.	Autopilot	Reaction Control	
8. Stop roll turn	M + 68 min.	CC&S	Autopilot and Reaction Control	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
9. Commence automatic reacquisition of celestial references (centering)		CC&S	Autopilot	
10. Canopus acquisition complete	M + 68 min.	--	--	
11. Verification of Canopus	M + 73 min.	SFOF		

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5.0B SECOND INTERPLANETARY TRAJECTORY CORRECTION				Correction may occur any time from 80 to 138 days
Same sequence as first interplanetary correction 5.0A. If done before 80 days	T < 80 days			
Same as third interplanetary correction if done after 80 days	T >80 days			



4.0(B) INTERPLANETARY CRUISE (SECOND)
4.1(B) RETURN TO CRUISE MODE
4.2(B) COAST IN TRAJECTORY
4.3(B) UPDATE CANOPUS ANGLE
4.4(B) TRANSMIT ENGINEERING AND SCIENCE DATA
4.5(B) SWITCH FROM OMNI TO HIGH-GAIN ANTENNA
4.6(B) UPDATE HIGH-GAIN ANTENNA POSITION
4.7(B) CHANGE DATA TRANSMIT BIT RATES
4.8(B) RECALIBRATE SCIENCE ELEMENTS AS REQUIRED
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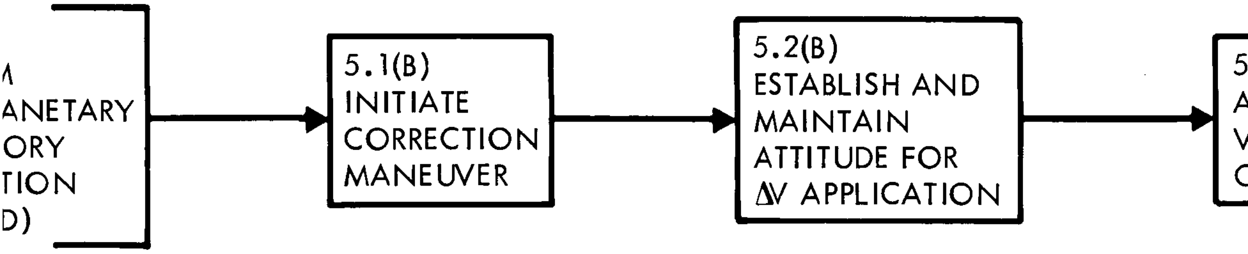

Figure 3.

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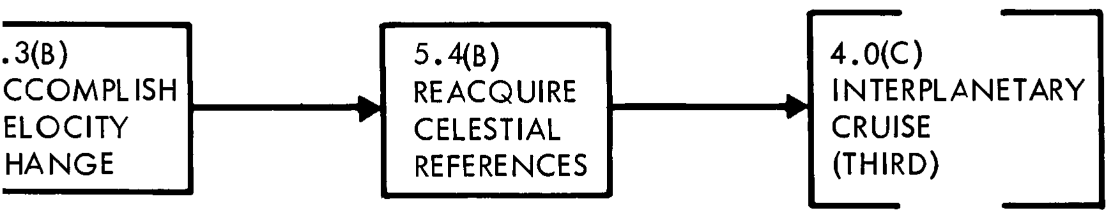


9-9: 1971 Mission — 4.0 (B) Interplanetary Cruise (Second) and
5.0 (B) Interplanetary Trajectory Correction (Second)

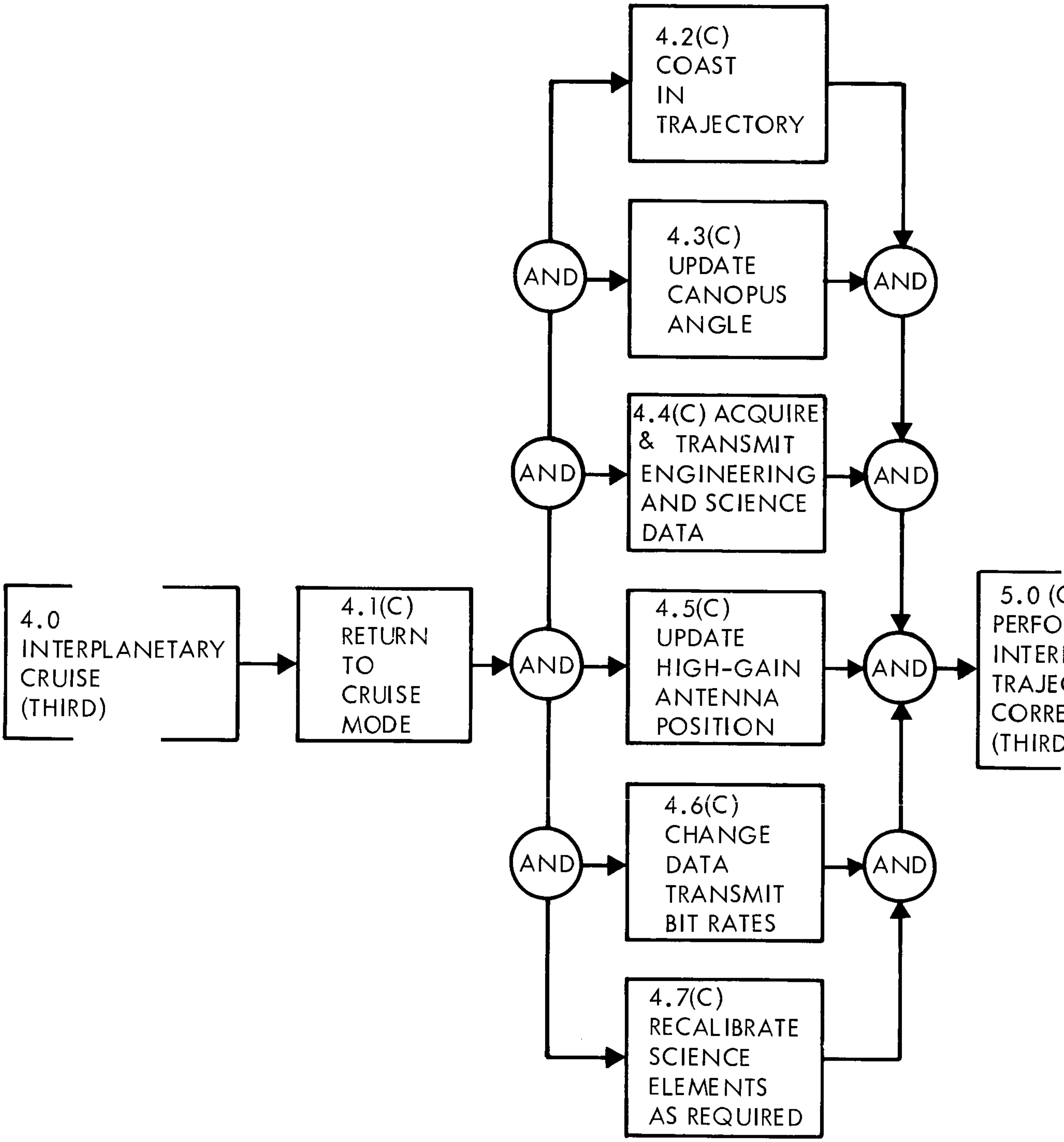
3

FLIGHT SEQUENCE

	EVENT	TIME	SOURCE	DESTINATION	COMMENTS
4.0C	INTERPLANETARY CRUISE (THIRD)	T+80 to 200 days			
4.1C	RETURN TO CRUISE MODE (SAME SEQUENCE AS 4.1A)				
4.2C	COAST IN TRAJECTORY				
	During third coast functions 4.3C, 4.4C, and 4.5C occur				
4.3C	UPDATE CANOPUS CONE ANGLE	T+172 days	CC&S	Attitude Control	CC&S Stored Command
	Set Canopus Cone Angle No. 3				
4.4C	ACQUIRE & TRANSMIT ENGINEERING AND SCIENCE DATA (SAME AS 4.4A and 4.5B)		--	--	Data Mode 2; 2/3 science and 1/2 engineering
4.5C	UPDATE HIGH GAIN ANTENNA POSITION (SAME AS 4.6B)		CC&S	Mechanisms	
4.6C	CHANGE DATA TRANSMIT BIT RATES		CC&S	Telecom	
4.7C	RECALIBRATE SCIENCE ELEMENTS AS REQUIRED		CC&S	Science	
5.0C	INTERPLANETARY TRAJECTORY CORRECTION (THIRD) (Same as first correction except that communication is via high-gain antenna. This requires reacquisition of Earth following roll	T+200 days			

FLIGHT SEQUENCE

	EVENT	TIME	SOURCE	DESTINATION	COMMENTS
	maneuvers, Antenna reposition commands must be sent and stored. Verification time is increased due to the increased signal transmission time.)				
4.0D	INTERPLANETARY CRUISE (FOURTH)				
4.1D	RETURN TO CRUISE MODE (SAME SEQUENCE AS 4.1A)	T + 200 to 205 days			
4.2D	COAST IN TRAJECTORY				
	During fourth coast functions 4.3D, 4.4D, and 4.5D occur				
4.3D	UPDATE CANOPUS CONE ANGLE (IF REQUIRED)	CC&S	CC&S	Attitude Control	CC&S Stored Command
4.4D	ACQUIRE AND TRANSMIT ENGINEERING AND SCIENCE DATA (SAME AS 4.4A AND 4.5A)				Data Mode 2; 2/3 science and 1/3 engineering
4.5D	UPDATE HIGH GAIN ANTENNA POSITION (SAME AS 4.6B)				
4.6D	CHANGE DATA TRANSMIT BIT RATES		CC&S	Telecom	
4.7D	RECALIBRATE SCIENCE ELEMENTS AS REQUIRED		CC&S	Telecom	


4.0 INTERPLANETARY CRUISE (THIRD)
4.1(C) RETURN TO CRUISE MODE
AND
4.2(C) COAST IN TRAJECTORY
4.3(C) UPDATE CANOPUS ANGLE
4.4(C) ACQUIRE & TRANSMIT ENGINEERING AND SCIENCE DATA
4.5(C) UPDATE HIGH-GAIN ANTENNA POSITION
4.6(C) CHANGE DATA TRANSMIT BIT RATES
4.7(C) RECALIBRATE SCIENCE ELEMENTS AS REQUIRED
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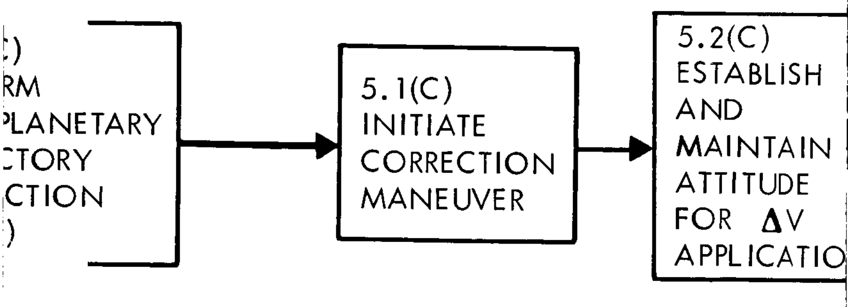
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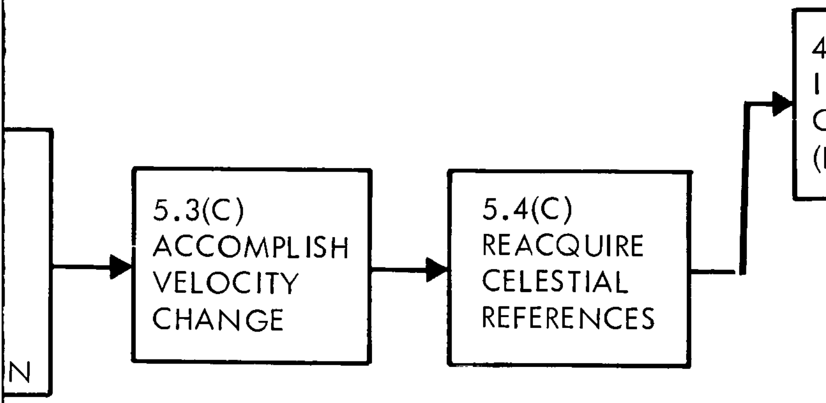

5.3(C)
ACCOMPLISH
VELOCITY
CHANGE
5.4(C)
REACQUIRE
CELESTIAL
REFERENCES

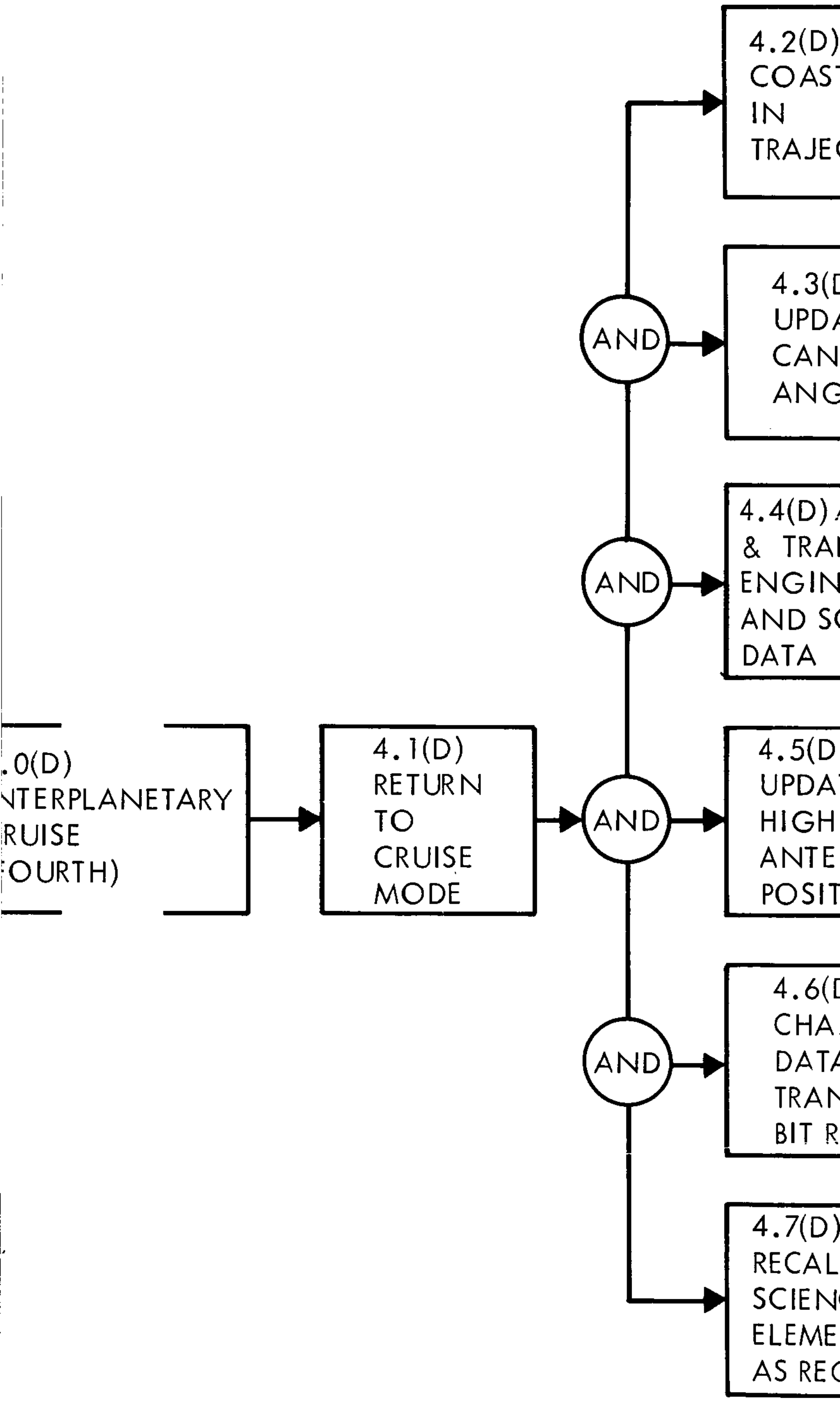
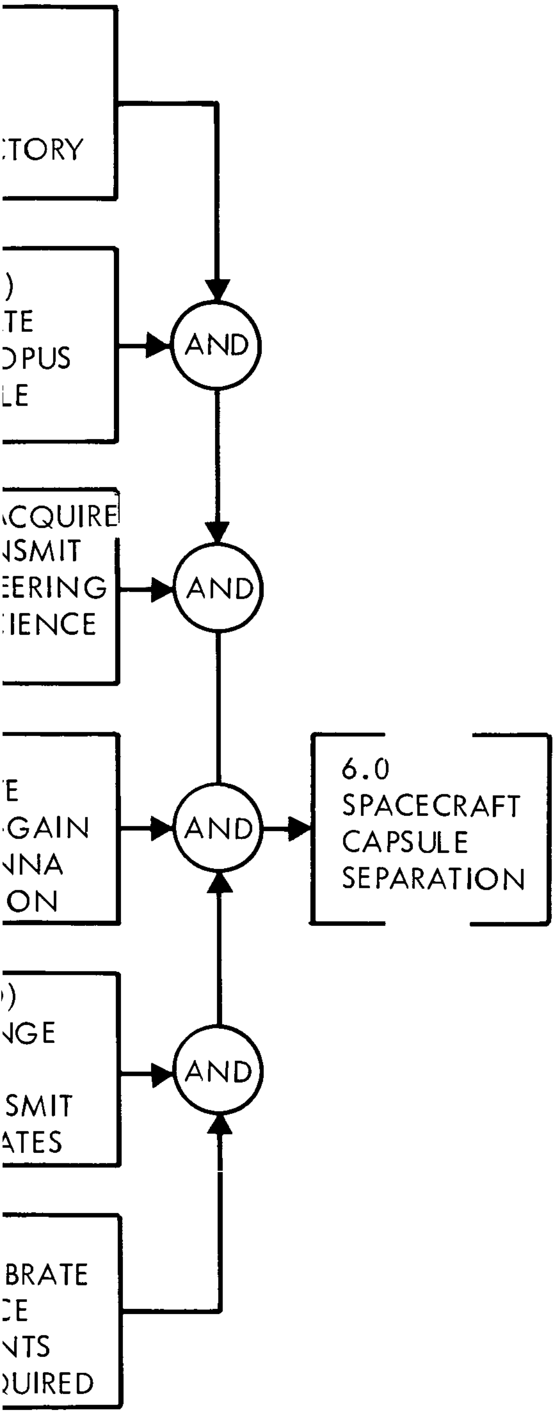


Figure 3.9-10: 1971 Mission — 4.0(C)
5.0(C) Interplanetary T
and 4.0(D) Interplaneta

203
(4)



Interplanetary Cruise (Third),
rajectory Correction (Third),
ry Cruise (Fourth)

FLIGHT SEQUENCE

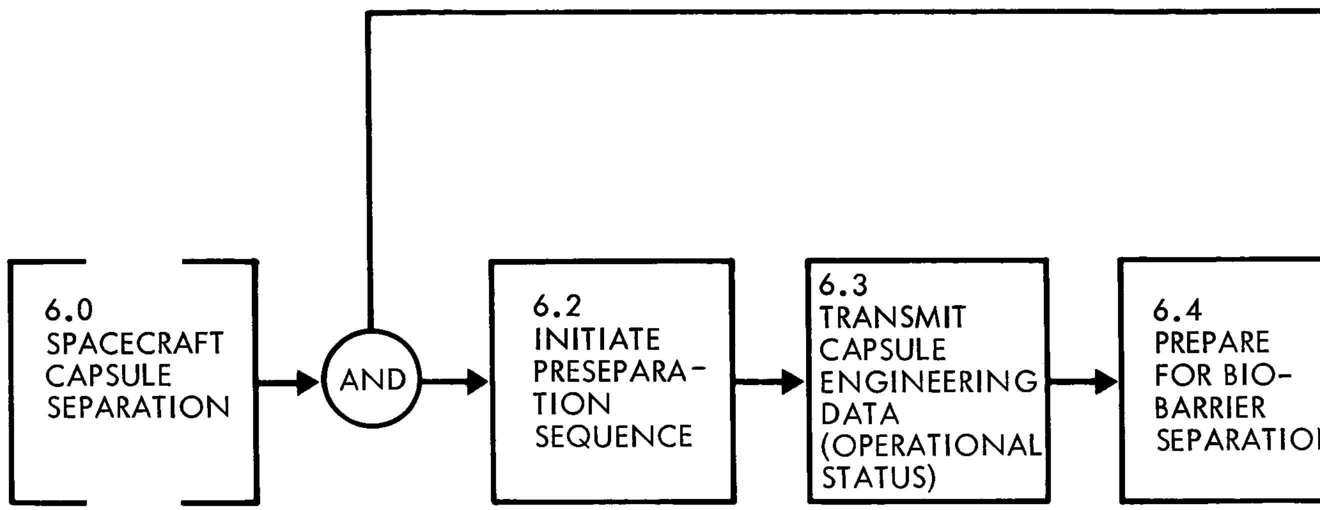
EVENT	TIME	SOURCE	DESTINATION	COMMENTS
6.0 SPACECRAFT CAPSULE SEPARATION	T + 205 days	DSN	CC&S	35 minutes required for separation sequence
6.1 PROVIDE SPACECRAFT (S/C) POWER TO CAPSULE--PROVIDE UP TO 50 WATTS UNTIL 6.9		Power	Capsule	
6.2 INITIATE PRESEPARATION SEQUENCE		DSN	CC&S	
1. Turn off all cruise science		CC&S	Science	
2. Switch to engineering data transmission only. Data Mode 1		CC&S	Telecom	
3. Load Capsule Programmer		CC&S	Capsule	
4. Data tape recorder on		CC&S	DHS	
5. Activate capsule sequence timer		CC&S	Capsule	
6. Synchronize S/C and capsule timer		CC&S	Capsule	
7. Verify capsule programmer ready		Capsule	CC&S	
8. Activate capsule entry science		CC&S	Capsule	
9. Activate capsule telecommunications and establish VHF radio link with S/C		CC&S	VHF (capsule) VHF (S/C)	
10. Select data link		CC&S	Capsule	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
11. Switch on detector and receiver A or B		CC&S	VHF (S/C)	
6.3 TRANSMIT CAPSULE ENGINEERING DATA (OPERATIONAL STATUS)		CC&S	Telecom	
1. Initiate capsule checkout		CC&S	Capsule	
2. Conduct capsule checkout and relay to S/C		Capsule	S/C	
3. Transmit engineering data to define operational status of capsule		Telecom	DSN	
4. Receive go-ahead for separation	S = 0	DSN	Telecom/ CC&S	
6.4 PREPARE FOR BIO-BARRIER SEPARATION				
Transmit signal to remove capsule separation inhibit		CC&S	Pyro	
6.5 COMMAND BIO-BARRIER SEPARATION		CC&S	Pyro	Signal to fire squibs
6.6 SEPARATE BIO-BARRIER AND VERIFY		Pyro	CC&S	Forward end of canister
6.7 ORIENT S/C CAPSULE SEPARATION AXIS				
1. Receive and Store:		DSN	Telecom and CC&S	
a. Roll turn magnitude and direction				

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
b. Yaw turn magnitude and direction. c. Maneuver start time d. Antenna repositioning				Backup shutoff in case of accelerometer malfunction
2. Switch to data Mode No. 1		CC&S	Telecom	
3. Switch gyros to rate control		CC&S	Attitude Ref. via Autopilot	
4. Switch autopilot to gyro reference		CC&S	Autopilot	
5. Decrease dead zone		CC&S	Autopilot	
6. Command roll maneuver		CC&S	Autopilot	
7. Start roll		Autopilot	Reaction Control	
8. Stop roll		Autopilot	Reaction Control	
9. Re-acquire Earth		CC&S	Autopilot & Reaction Control	
10. Verification of roll		CC&S	Mechanisms	Antenna positioning
a. Transmit degrees of roll to SFOF		CC&S/ Telecom	SFOF	
b. Ground evaluation		SFOF		
c. Receive roll verified, proceed with yaw		SFOF	Telecom/ CC&S	


6.0 SPACECRAFT CAPSULE SEPARATION
AND
6.2 INITIATE PRESEPARA-TION SEQUENCE
6.3 TRANSMIT CAPSULE ENGINEERING DATA (OPERATIONAL STATUS)
6.4 PREPARE FOR BIO-BARRIER SEPARATIO

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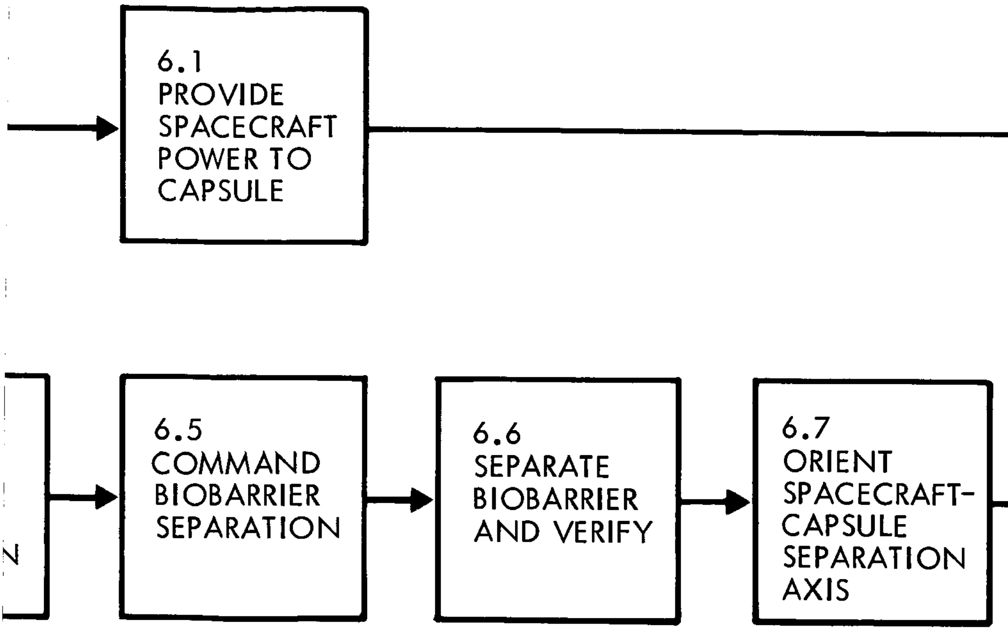
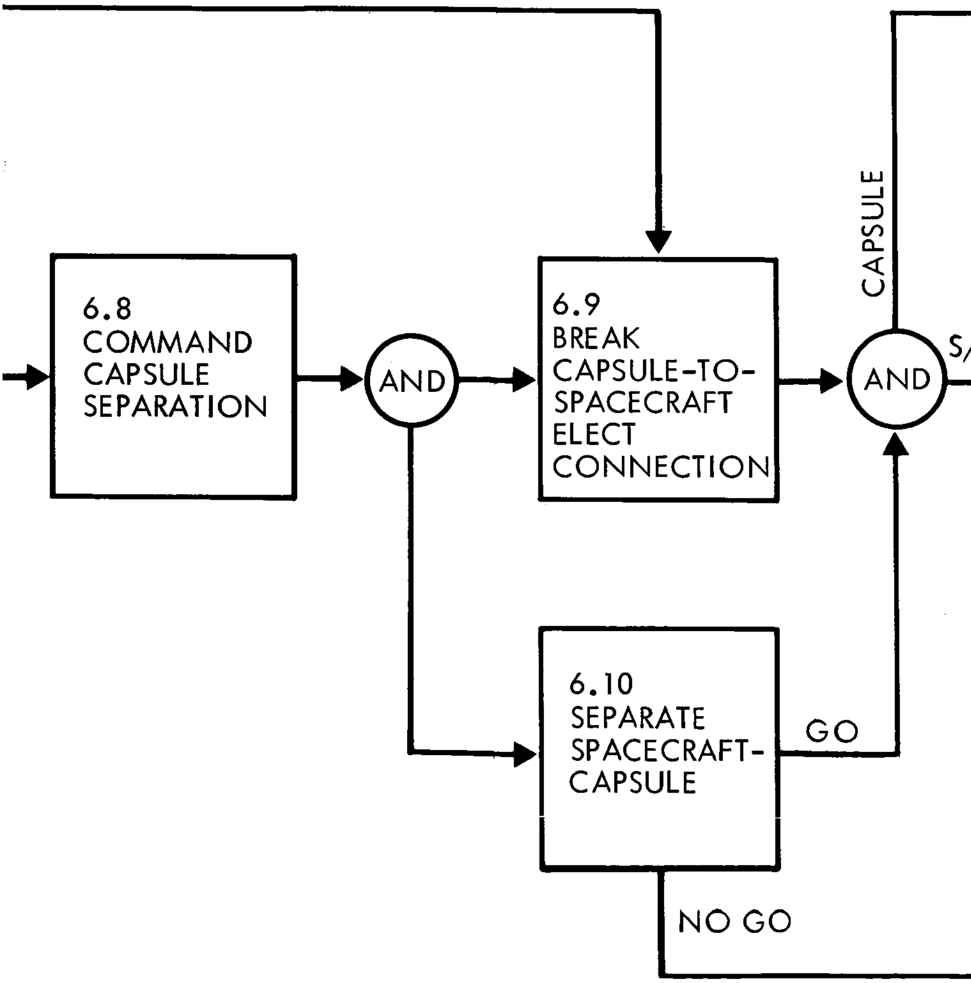

6.1 PROVIDE SPACECRAFT POWER TO CAPSULE
N
6.5 COMMAND BIOBARRIER SEPARATION
6.6 SEPARATE BIOBARRIER AND VERIFY
6.7 ORIENT SPACECRAFT-CAPSULE SEPARATION AXIS


6.8
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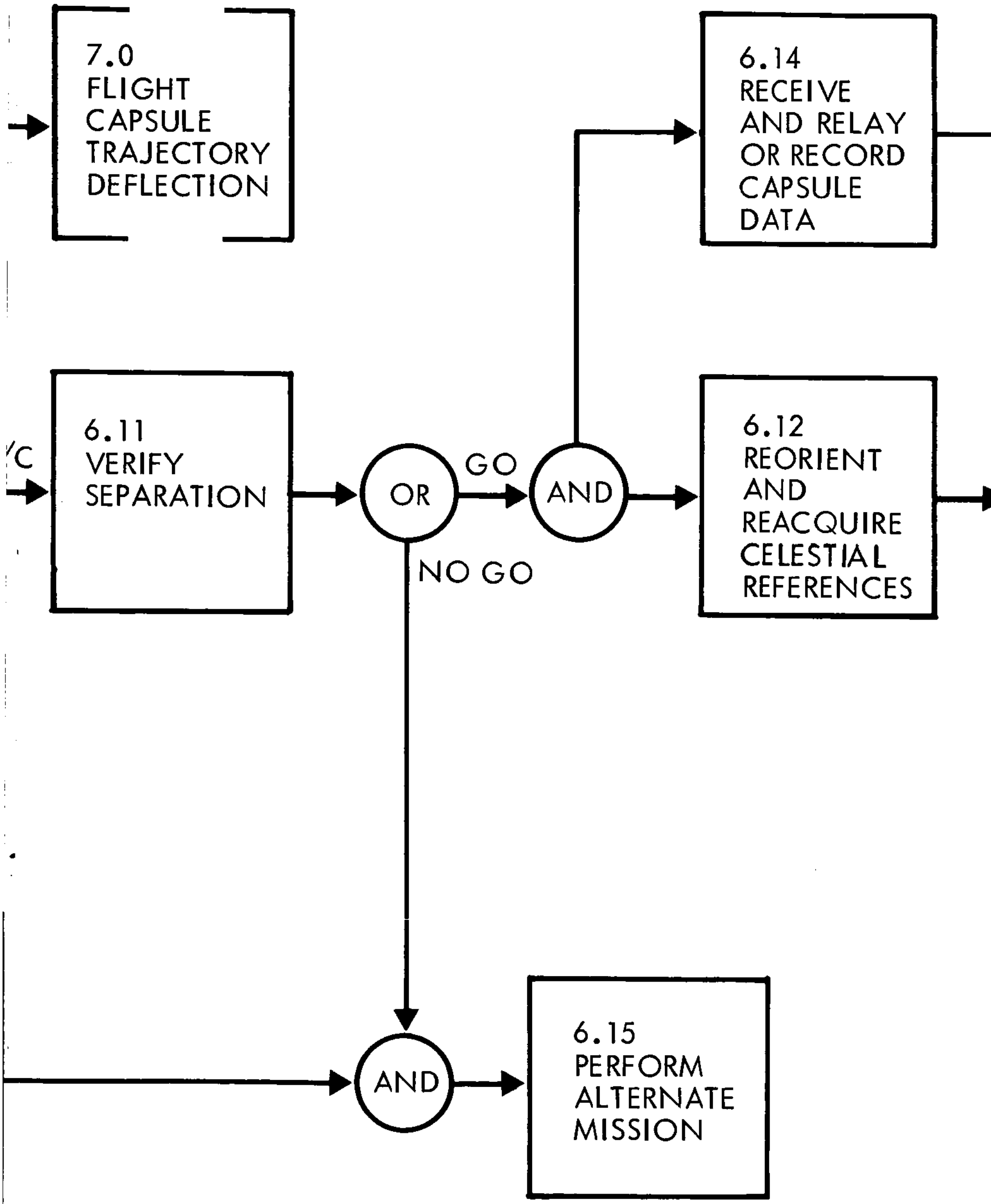
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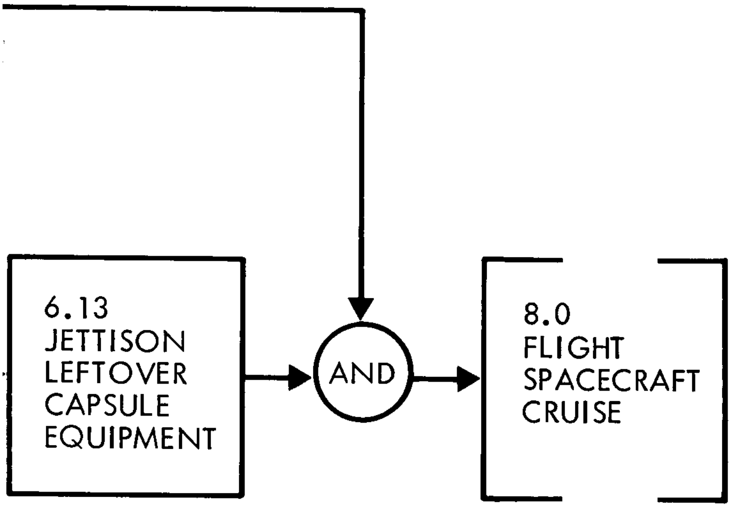


e 3.9-11: 1971 Mission — 6.0 Spacecraft Capsule Separation

5

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
8.0 FLIGHT SPACECRAFT CRUISE	T+206 to 214 days			Following capsule separation
8.1 ESTABLISH FLIGHT SPACECRAFT CRUISE MODE				
1. Switch to solar panel power		Automatic	Power	
2. Switch to automatic reacquisition of celestial references		CC&S	Autopilot	
3. Switch to data mode 2		CC&S	Telecom	
4. Turn battery charger on		Automatic	Power	
5. Turn cruise science on		CC&S	Science	
6. Turn on data recorder		CC&S	Telecom	
7. Set Canopus cone angle as required		CC&S	Star Tracker	
8. Set antenna step as required		CC&S	Mechanisms	Once every day
9. Receive and store, or transmit, data (VHF) from capsule		Capsule	Telecom	
8.2 RECEIVE ORBIT INSERTION PARAMETERS	T+206 to 208 days			
1. Receive and store updated commands for encounter		DSN	Telecom & CC&S	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
2. Receive and store orbit injection parameters		DSN	Telecom & CC&S	Telecommunications receives and transmits to CC&S for storage
a. Roll turn magnitude and direction b. Yaw turn magnitude and direction c. Pitch turn magnitude and direction (if required) d. Propulsion ignition time e. Antenna positioning f. Maneuver start time				
8.3 TERMINATE CRUISE SCIENCE MODE	T + 213 days			
1. Switch off all cruise science		CC&S	Science	
2. Switch to data mode 1		CC&S	Telecom	
8.4 BEGIN RECEIVING ENCOUNTER SCIENCE MODE	T + 214 days			
1. Unlatch SCAN platform		CC&S	Pyro	
2. Drive SCAN platform to operating position		CC&S	Mechanism	
3. Provide out and lock signal		Mechanism	Telecon	
4. Switch on data recorder		CC&S	Telecom	
8.5 RECORD SCIENCE DATA (Begin receiving capsule entry data)				
1. Receive and record capsule science		Capsule	Telecom	
2. Receive and record encounter science data		Science	Telecom	

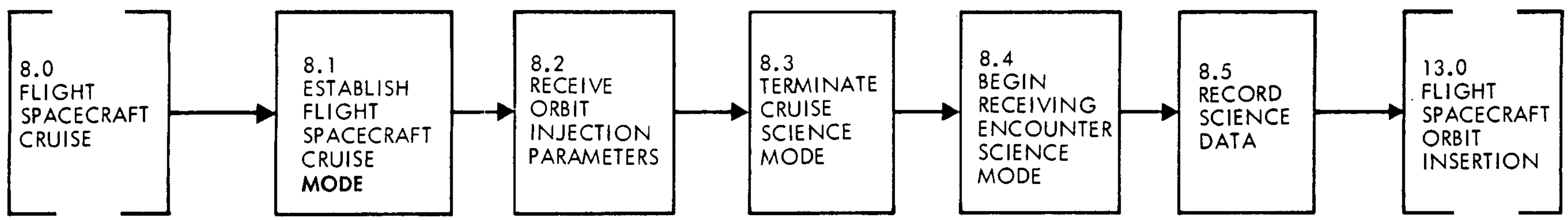


Figure 3.9-12: 1971 Mission — 8.0 Flight Spacecraft Cruise Towards Mars — Gather Data

FLIGHT SEQUENCE

	EVENT	TIME	SOURCE	DESTINATION	COMMENTS
13.1	RECEIVE UPDATED ORBIT PARAMETERS AS REQUIRED	T+214 days	DSN	Telecom and CC&S	
	Receive and Store				
	1. Roll turn magnitude and direction				
	2. Yaw turn magnitude and direction				
	3. Pitch turn magnitude and direction (if required)				
	4. Propulsion ignition time				
	5. Antenna position				
	6. Maneuver start time				
13.2	TERMINATE FINAL CRUISE MODE	T+214 days			
	1. Celestial references unlocked		CC&S	Attitude Control	
	2. Battery charger off				
	3. Cruise science data		CC&S	Science	
	4. Data recorder off		CC&S		
13.3	ORIENT FLIGHT SPACECRAFT TO INSERTION ATTITUDE	T+215 days			
	1. Command for maneuver start		DSN	Telecom and CC&S	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
2. Switch to data mode #1		CC&S	Telecom	
3. Switch gyros to rate control		CC&S	Attitude Ref. via Autopilot	
4. Switch autopilot to gyro reference		CC&S	Autopilot	
5. Remove propulsion inhibit and thrust vector control		CC&S	Pyro	
6. Decrease dead zone		CC&S	Autopilot	
7. Command roll maneuver		CC&S	Autopilot	
8. Start roll turn	I = 0	Autopilot	Reaction Control	I = Start of orbit insertion sequence
9. Stop roll turn	I + 2 min	CC&S	Autopilot & Reaction Control	
10. Reacquisition of Earth				
a. Antenna repositioning	I + 3	CC&S	Mechanisms	
b. Transit degrees of roll to Earth	I + 15	Telecom	SFOF	7 min trans time lag 5 min lock-in time

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
11. Verification of roll				
a. Ground evaluation	I + 20 min	SFOF		5 min Evaluation time
b. Receive roll verified, proceed with yaw	I + 27 min	SFOF	Telecom/ CC&S	7 min transmission time lag
12. Command yaw maneuver start	I + 27 min	CC&S	Autopilot	
13. Start yaw turn		CC&S	Autopilot	
14. Stop yaw turn	I + 34 min	CC&S	Autopilot & Reaction Control	
15. Verification of yaw	I + 53 min	S/C	SFOF	Two-way communication time lag plus ground evaluation time
16. Increase dead zone		CC&S	Autopilot	
13.4 PROVIDE THRUST				
1. Turn on accelerometer		CC&S	Autopilot	
2. Ignite retroengine igniter	I + 52 min	CC&S	Pyro	
3. Count accelerometer pulses		Accelerometer	CC&S	
4. Wait for engine cutoff	I + 55 min			

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
13.5 REORIENT SPACECRAFT				
1. Reacquire Earth (after burn)	1 I + 72	CC&S	Mechanism	1 17 minutes includes 7 minutes one-way communication time & 10 minutes for lockup procedures
2. Decrease dead zone		CC&S	Autopilot	
3. Command yaw maneuver		CC&S	Autopilot	
4. Start yaw turn	I + 72	Autopilot	Reaction Control	
5. Stop yaw turn	I + 79	CC&S	Autopilot	
6. Reacquire Sun	I + 80	Sensors	Reaction Control	
7. Command roll turn		CC&S	Autopilot	
8. Start roll turn	I + 80	Autopilot	Reaction Control	
9. Stop roll turn (canopus acquisition)	I + 82	Autopilot	Reaction Control	
10. Reacquire Earth	I + 95	CC&S/ Telecom	SFOF	Includes 1 minute antenna slew,7 min. one-way

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
				communication time & 5 minutes for lock-up procedures
11. Verify canopus				
a. Transmit sensor output signal to SFOF	I + 102	CC&S/ Telecom	SFOF	
b. Ground evaluation	I + 107	SFOF		
c. Receive canopus acquisition verified	I + 114	SFOF	Telecom/ CC&S	

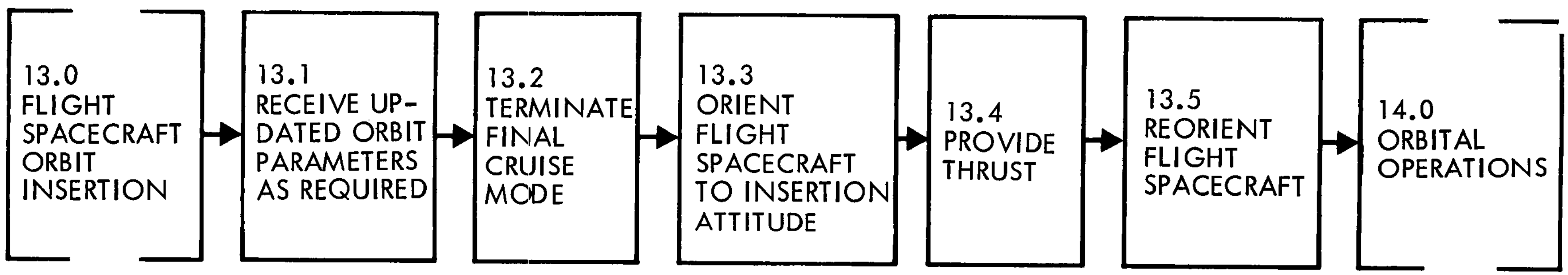


Figure 3.9-13: 1971 Mission — 13.0 Flight Spacecraft Insertion Into Mars Orbit

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
14.0 ORBITAL OPERATIONS				
14.1 ESTABLISH ORBIT PARAMETERS	T + 215 days			Two or more Mars orbits required
14.2 COAST IN ORBIT MAINTAIN CELESTIAL REFERENCES				Maximum orbit operation period = 6 months
1. Update high-gain antenna position		CC&S	Mechanisms	
2. Update canopus cone angle		CC&S	Mechanisms	
3. Maintain solar lock within tol-erance and reacquire after occulation		Autopilot	Reaction Control	Occultations
4. Maintain canopus lock and require after occulations		Autopilot	Reaction Control	
14.3 POSITION SCAN PLATFORM				
1. Deploy cover assembly and release		CC&S	Mechanisms	
2. Position scan platform		CC&S	Mechanisms	
3. Receive Time to record		DSN	CC&S	After orbit is established the time to initiate read-out of the recorded data to the telemetry system is provided by ground command.

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
4. Receive scanner instrument selection commands		DSN	CC&S/ Science	
14.4 ACQUIRE SCIENCE DATA				
1. Select data mode		CC&S	Teleccm	The actual orbital sequence is not defined at present
2. Turn science instruments on		CC&S	Science	
14.5 RECORD AND TRANSMIT SCIENCE DATA AT PROGRAMMED INTERVALS				
1. Select recorded science readout to T/M		DAS	Telecom	
2. Stop recorded science readout to Telecom and select data mode		CC&S	DAS	The actual orbital sequence is not defined at present
3. Transmit data to Earth		Telecom	DSN	
14.6 RECEIVE AND RECORD CAPSULE DATA		DSN	CC&S	
1. Receive command to initiate data receipt from capsule		DSN	Telecom	
2. Activate VHF radio unit		CC&S	Telecom	
3. Activate S/C data storage subsystem		CC&S	Telecom	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
14.7 TRIM ORBIT IF REQUIRED				
1. Receive command to initiate maneuver	218 days	DSN	CC&S	
2. Initiate sequence		CC&S	Reaction Control	
a. Position S/C for application of ΔV b. Activate TVC c. Apply thrust d. Terminate thrust e. Reacquire references				Sequence same as 3rd Midcourse 5. OC.
3. Check orbit parameters		DSN		
14.8 REPEAT SCIENCE DATA ACQUISITION SEQUENCE FOR EACH ORBIT				Repeat cycle of position of scan platform, acquisition and transmission of science and engineering data. Function: 14.2 thru 14.6
14.9 SWITCH TO OCCULTATION MODE	T + 255	Power	CC&S	Solar occultations will begin occurring after 40 days in orbit
1. Sun presence lost				
2. Change to inertial reference in pitch and yaw		CC&S	Autopilot	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINA-TION	COMMENTS
3. Sun presence		Power	CC&S	
4. Reacquire celestial refs.		CC&S	Autopilot	
14.10 TERMINATE MISSION	T + 395 days			

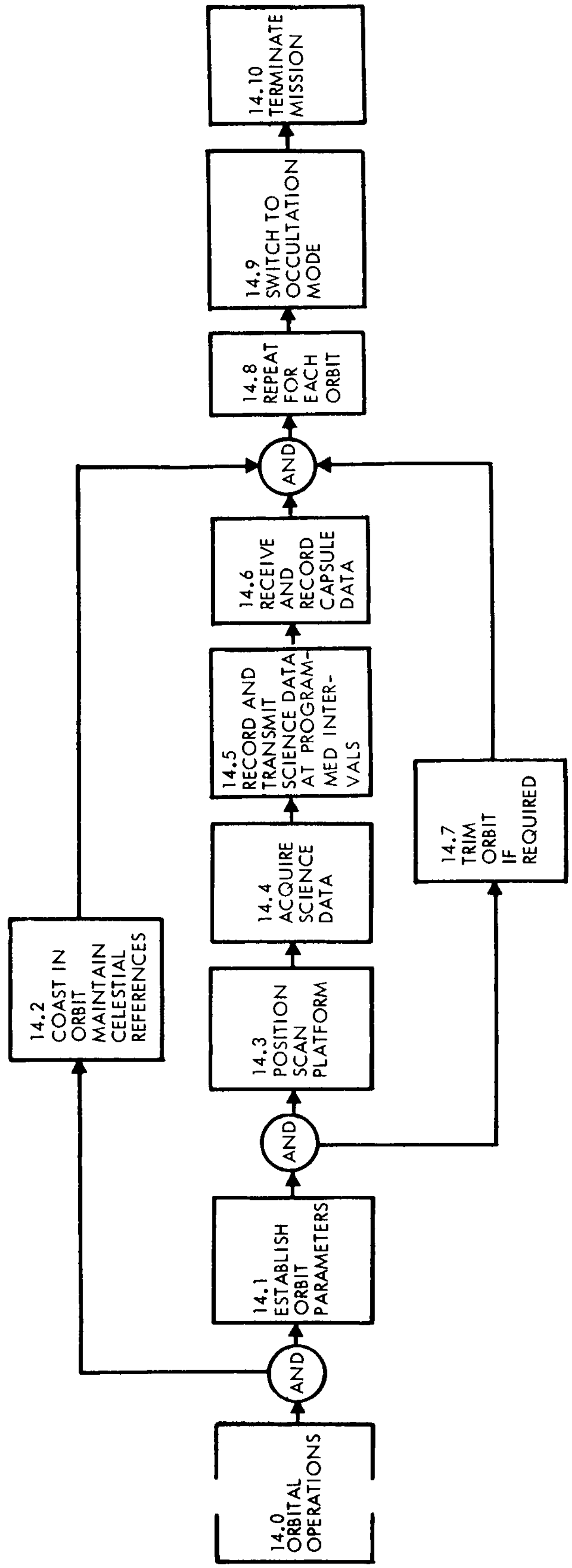


Figure 3.9-14: 1971 Mission — 14.0 Orbital Operations

3.9.3 Automatic Maneuver Sequence

Typical alternate 2 axis automatic mode maneuver sequence possible by reprogramming the CC&S are shown in the following pages. A relationship of ground transmitted commends for each sequence is as follows:

	Nominal Sequence Ground Commands			Automatic Sequence Ground Commands		
Maneuver Sequence	5.0 A First Mid-Course	5.0 C Third Mid-Course	13 Orbit Injection	5.0 A First Mid-Course	5.0 C Third Mid-Course	13 Orbit Injection
Correction Parameters	5	7	5	5	7	5
Orient Spacecraft	2	2	2	0	0	0
Velocity Maneuver	1	1	1	0	0	1
Reverse Spacecraft Orientation	1	1	1	0	0	0
Celestial Acquisition	1	1	1	0	0	0
	10	12	10	5	7	6
Total Necessary Ground Commands		32			18	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5.0A FIRST MIDCOURSE CORRECTION				
5.1A INITIATE CORRECTION MANEUVER	T + 5 days	DSN	Telecom & CC&S	
1. Receive and store - DSN to CC&S a. Line to begin maneuver sequence (T_1). b. Roll turn magnitude and direction. c. Yaw turn magnitude and direction. d. Midcourse burn time (backup) e. Velocity increment (V)				
5.2A ESTABLISH AND MAINTAIN ATTITUDE FOR V APPLICATION				
1. At (T_1) switch telemetry to data mode 1.	M = 0	CC&S	Telecom	M = Start Maneuver
2. Decrease dead zone.		CC&S	Autopilot	
3. Wait time for S/C attitude settling.				
4. Command roll maneuver. a. Select slew mode roll axis.	M + 1	CC&S	Autopilot	Autopilot automatic
5. Complete roll maneuver. a. Switch to inertial hold mode roll axis.	M + 4	CC&S	Autopilot	Autopilot automatic
b. Transmit degrees of roll to SFOF automatically.		CC&S	Telecom	

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
6. Wait time S/C attitude settling				
7. Command yaw maneuver	M + 5	CC&S	Autopilot	
a. Select slew mode yaw axis				Autopilot automatic
b. Switch to inertial hold mode pitch axis.				Autopilot automatic
8. Complete yaw maneuver.	M + 20	CC&S	Autopilot	
a. Switch to inertial hold mode yaw axis.				Autopilot automatic
b. Transmit degrees of yaw to SFOF automatically		CC&S	Telecom	
9. Wait time S/C attitude settling and for ground interrupt in case maneuver sequence is faulty.				
5.3A ACCOMPLISH VELOCITY CHANGE				
1. Turn on accelerometer.	M + 25	CC&S	Att. Ref.	
2. Arm the monopropellant engine nitrogen and fuel squibs.		CC&S	Propulsion	
3. Backup arm the monopropellant engine nitrogen and fuel squibs. Select midcourse motors used (1 + 3) or (2 + 4).		CC&S	Propulsion	
4. Ignite midcourse motor and select midcourse thrust vector control mode.	M + 25	CC&S	Propulsion & autopilot	

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5. Count accelerometer pulses.		Att. Ref.	CC&S	
6. Shut off midcourse motor at (ΔV) and terminate midcourse thrust vector control mode.	M + 35	Att. Ref. CC&S	Propulsion & autopilot	
7. Wait S/C attitude control settling time.				
a. Transmit ΔV to SFOF (automatic)		CC&S	Telecom	
8. Disarm the monopropellant engine-nitrogen and fuel squibs.		CC&S	Propulsion	
9. Backup disarm monopropellant engine-nitrogen and fuel squibs.		CC&S	Propulsion	
5.4A REORIENT SPACECRAFT				
1. Command reverse yaw maneuver	M + 35	CC&S	Autopilot	
a. Select slew mode yaw axis				Autopilot automatic
2. Complete yaw maneuver	M + 50	CC&S	Autopilot	
a. Switch yaw gyro to position mode.				Autopilot automatic
b. Telemeter degrees of yaw maneuver (automatic)		CC&S	Telecom	
3. Wait time S/C attitude settling				
4. Command reverse roll maneuver.	M + 51	CC&S	Autopilot	
a. Select slew mode roll axis				Autopilot automatic

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5. Complete roll maneuver	M + 54	CC&S	Autopilot	
a. Switch to inertial hold mode roll axis				Autopilot automatic
5.5A REACQUIRE CELESTIAL REFERENCES				
1. Command Sun acquisition mode (centering automatic)	M + 54	CC&S	Autopilot	
2. Wait line for Sun acquisition				
3. Command1Canopus acquisition mode	M + 55	CC&S	Autopilot	
4. Wait time for ground intervention if Canopus not acquired.				
a. Transmit Sun and Canopus presence to SFOF.		CC&S	Telecom	
5. Increase dead zone.	M + 56	CC&S	Autopilot	

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5.0C THIRD MIDCOURSE TRAJECTORY CORRECTION				
5.1C INITIATE CORRECTION MANEUVER				
1. Receive and store - DSN to CC&S a. Time to begin Maneuver Sequence (T_1) b. Roll turn magnitude and direction. c. Yaw turn magnitude and direction. d. Midcourse burntime (backup) e. Velocity increment (ΔV) f. Antenna right-left change g. Antenna up-down change		DSN	Telecom & CC&S	
5.2C ESTABLISH AND MAINTAIN ATTITUDE FOR ΔV APPLICATION				
1. At (T_1) switch telemetry to data mode 1, record	M = 0	CC&S	Telecom	M = start maneuver
2. Decrease dead zone.		CC&S	Autopilot	
3. Wait time for S/C attitude settling.				
4. Command roll maneuver. a. Select slew mode roll axis	M = 1	CC&S	Autopilot Reaction Control	Autopilot automatic

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5. Stop roll maneuver	M + 4 min	CC&S	Autopilot	
a. Switch to inertial hold mode roll axis.				Autopilot automatic
6. Wait time for S/C attitude settling.				
7. Command yaw maneuver	M + 5 min	CC&S	Autopilot Reaction Control	
a. Select slew mode yaw axis				Autopilot automatic
b. Switch to inertial hold mode pitch axis.				Autopilot automatic
8. Complete yaw maneuver	M + 20	Autopilot	Reaction Control	
a. Switch to inertial hold mode yaw axis.				Autopilot automatic
9. Command antenna position - right or left, up or down	M + 21	CC&S	Mechanisms	
10. Switch telemetry to mode 3 playback.		CC&S	Telecom	
a. Transmit all telemetry data to SFOF including roll and yaw degrees of maneuver		Telecom	SFOF	
11. Wait time S/C attitude settling and for ground interrupt in case maneuver sequences are faulty.	M + 40	--	--	7 min communication one way 2-way communication required. 5 min ground evaluation

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINA-TION	COMMENTS
5.3C ACCOMPLISH VELOCITY CHANGE				
1. Turn on accelerometer.		CC&S	Att. Ref.	
2. Arm the monopropellant engine-nitrogen and fuel squibs.		CC&S	Pyro	
3. Backup arm the monopropellant engine-nitrogen and fuel squibs.		CC&S	Pyro	
4. Select midcourse motors (1 & 3) or (2 & 4).		CC&S	CC&S	
5. Ignite midcourse motor & select midcourse thrust vector control mode.	M + 40	CC&S	Propulsion & Autopilot	
6. Count accelerometer pulses.		Att. Ref.	CC&S	
7. Shut off midcourse motor at ΔV and terminate midcourse thrust vector control mode.	M + 50	CC&S	Propulsion & Autopilot	
8. Wait S/C attitude control settling time.				
a. Transmit ΔV to SFOF(automatic)		CC&S	Telecom	
9. Disarm the monopropellant engine-nitrogen and fuel squibs.	M + 51	CC&S	Pyro	
10. Backup disarm the monopropellant engine-nitrogen and fuel squibs.		CC&S	Pyro	

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
5.4C REORIENT SPACECRAFT				
1. Switch telemetry to mode 1 engineering data record.		CC&S	Telecom	
2. Command reverse yaw maneuver.	M + 51	CC&S	Autopilot	
a. Select slew mode yaw axis				Autopilot automatic
3. Complete yaw maneuver	M + 66	CC&S	Autopilot	
a. Switch yaw gyro to position mode.				Autopilot automatic
4. Wait time S/C attitude settling.				
5. Command reverse roll maneuver.	M + 67	CC&S	Autopilot	
a. Select slew mode roll axis				Autopilot automatic
6. Complete roll maneuvers.	M + 70	CC&S	Autopilot	
a. Switch to inertial hold mode roll axis				Autopilot automatic
7. Command antenna position - left or right, down or up.		CC&S	Mechanisms	
8. Switch telemetry to mode 3 engineering data playback.	M + 71	CC&S	Telecom	
5.5C REACQUIRE CELESTIAL REFERENCES				
1. Command Sun acquisition mode.		CC&S	Autopilot	

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
2. Wait time for Sun acquisition.				
3. Command Canopus acquisition mode	M + 72	CC&S	Autopilot	
4. Wait time for ground intervention of Canopus not acquired. a. Transmit Sun and Canopus presence to SFOF.				
5. Increase dead zone.		CC&S	Autopilot	
6. Switch telemetry to mode 1 engineering data.	M + 73	CC&S	Telecom	

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
13.0 FLIGHT SPACECRAFT ORBIT INJECTION				
13.1 RECEIVE UPDATED ORBIT PARAMETERS				
Same as 5.1C except a. Midcourse burn time b. Velocity increment				
13.2 TERMINATE FINAL CRUISE MODE				
Same as paragraph 3.9.2 in nominal sequence				
13.3 ORIENT FLIGHT SPACECRAFT TO INSERTION ATTITUDE	I + 45			I = Start of insertion maneuver
Same as 5.2C				
13.4 ACCOMPLISH VELOCITY CHANGE				
1. Turn on accelerometer.		CC&S	Attitude Ref.	
2. Arm solid engine actuators nitorgen squibs.		CC&S	Pyro	
3. Backup arm solid engine jet vanes nitrogen squibs		CC&S	Pyro	

ALTERNATE SEQUENCE

FLIGHT SEQUENCE

EVENT	TIME	SOURCE	DESTINATION	COMMENTS
4. Wait time (10 Minutes) e	M + 45 to M + 55			Wait time terminated at M + 55 by '5' below.
a. Generate command for motor burn	M + 40	SFOF		
b. Receive Command	M + 47	SFOF	Telecom/ CC&S	
c. Ignite orbit insertion engine	M + 47	CC&S	Propulsion	
5. Disarm solid engine jet vanes nitrogen squibbs	M + 55	CC&S	Pyro	
13.5 REORIENT SPACECRAFT	I + 51			
Same as 5.4C				
13.6 REACQUIRE CELESTIAL REFERENCES	I + 56			
Same as 5.5C				

ALTERNATE SEQUENCE

3.10 VOYAGER FLIGHT EQUIPMENT, SPACECRAFT LAYOUT, AND CONFIGURATION

3.10.1 Scope

This section defines the Voyager spacecraft configuration and layout including: spacecraft reference axes and planes, coordinate system, mechanical alignment provisions, general arrangement of the exterior of the spacecraft, and equipment arrangement. A functional block diagram is included for additional reference.

The preferred configuration, Boeing Model 945-6026, was selected because it offers the following important features:

1) Convenient access to the electronic assemblies and propulsion subsystems, resulting in ease of installation, maintenance, and testing, thereby enhancing the spacecraft reliability;
2) A large (8 ft. by 12 ft.) paraboloidal antenna, allowing for telecommunication data rates consistent with real-time transmission;
3) Modular construction of subsystems, allowing complete checkout before installation in the spacecraft;
4) Versatility in the sizing, location, and construction of electronic assemblies, providing for optimum grouping of electronic functions for simple interfaces, and for testing and installation. Thermal balance and center-of-gravity location are more easily attained.
5) Versatility in general configuration for adaptability to 1973 and later Voyager missions. The Spacecraft Bus is so designed that a range of trajectories and Mars orbits can be achieved for the 1973 mission. Also, the 1975 and 1977 flyby missions can be

performed with the orbit insertion propulsion module removed, or if mission plans change so that orbital missions are desired for those years, the orbit insertion propulsion system can be included to give a range of achievable Mars orbits.

6) Adaptability of general configuration to the 1969 test flight. The configuration is compatible to the Atlas Centaur launch vehicle with only minimum changes (principally to the high-gain antenna and to the solar panels), thereby retaining a high degree of commonality to the 1971 configuration.

A summary of the alternate configurations that were studies, and the justification for the selection of this configuration is presented in D2-82709-2, Section 3.2.

3.10.2 Applicable Documentation and Drawings

3.10.2.1 Documents

M61-205, "Optical Alignment Reference Manual;"

M61-100, "Optical Tooling Instruments, Accessories and Techniques Manual;"

M41-02, "The Boeing Optical Tooling Manual;"

M61-200, "Optical Tooling Service Manual" (Volume I & II);

D1-8001, "The Boeing Company Measuring Program;"

D1-8002, "The Boeing Company Primary Standards Capability;"

D1-8005, "The Boeing Company Calibration Procedure;"

D2-5378, "Measurement Test Equipment Calibration Procedure" (Volume III).

3.10.2.2 Drawings

Figure 3.10-1, Isometric of Flight Spacecraft

Figure 3.10-2, Boeing Drawing 25-50034, Sheet 1-- General Arrangement;

Figure 3.10-3, Boeing Drawing 25-50034, Sheet 2-- Inboard Profile;

Figure 3.10-4, Boeing Drawing 25-50034, Sheet 5-- Functional Block Diagram;

Figure 3.10-5, Boeing Voyager Coordinate System.

3.10.3 Functional Description

3.10.3.1 General Description

The general arrangement of the spacecraft is shown in Figure 3.10-1 and 3.10-2 . This configuration complies with the Jet Propulsion Laboratory "Mission Specification" (Project Document 45) spacecraft envelope, capsule interface, and launch vehicle interface.

The pertinent features of the spacecraft are:

1) An equipment-support module that provides the mounting surface for the equipment packages.

2) A lower truss that supports the equipment-support module, propulsion system reaction-control module, and all deployable components.

3) Gross total solar-array area of 258 square feet, (236-square-foot net area) with growth potential to 375 square feet by adding one segment to each of the three arrays.

4) A single planet-scan platform built to accommodate the hypothetical science payload (5.2 cubic feet) described in JPL Document No. 46, "Mission Guidelines".

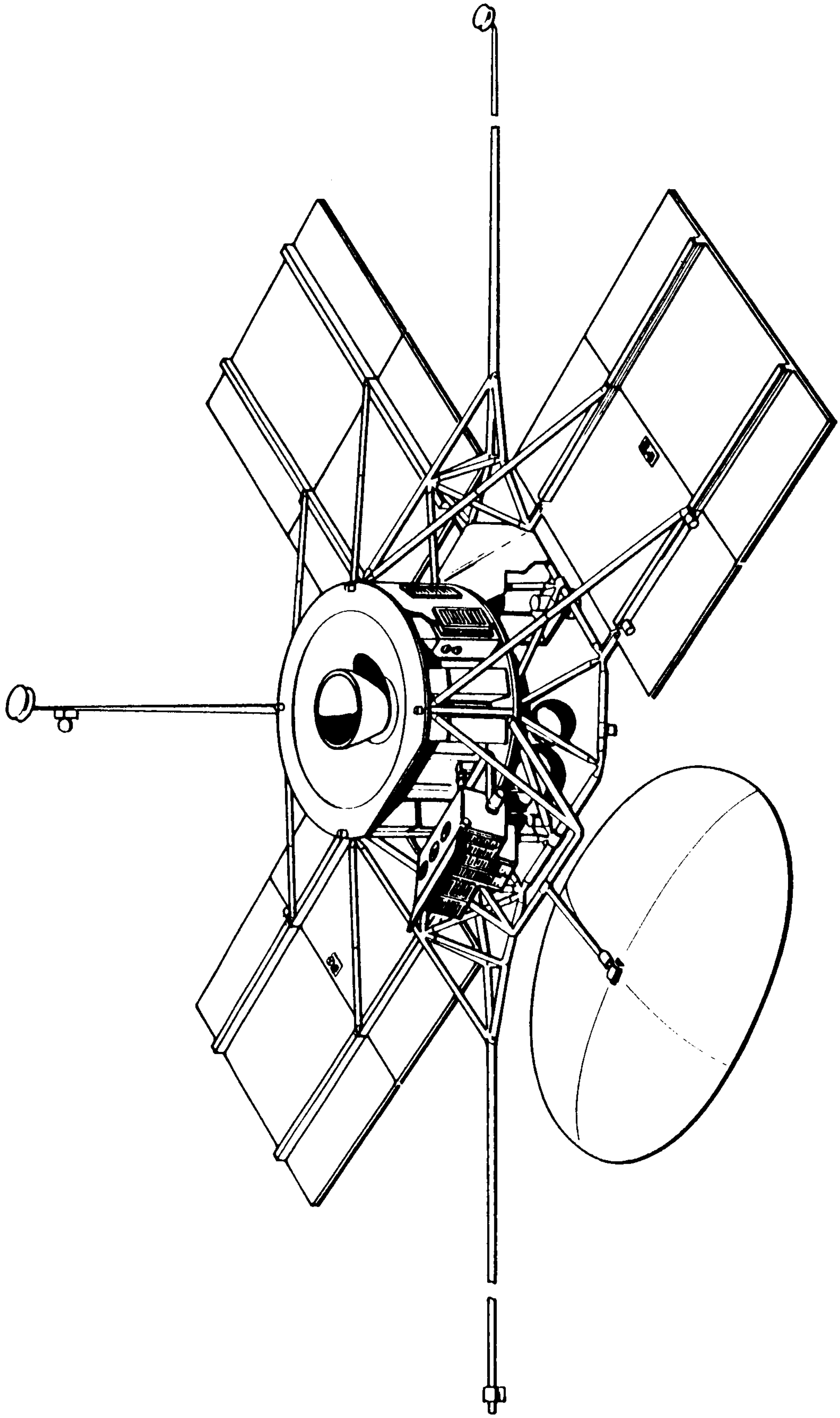
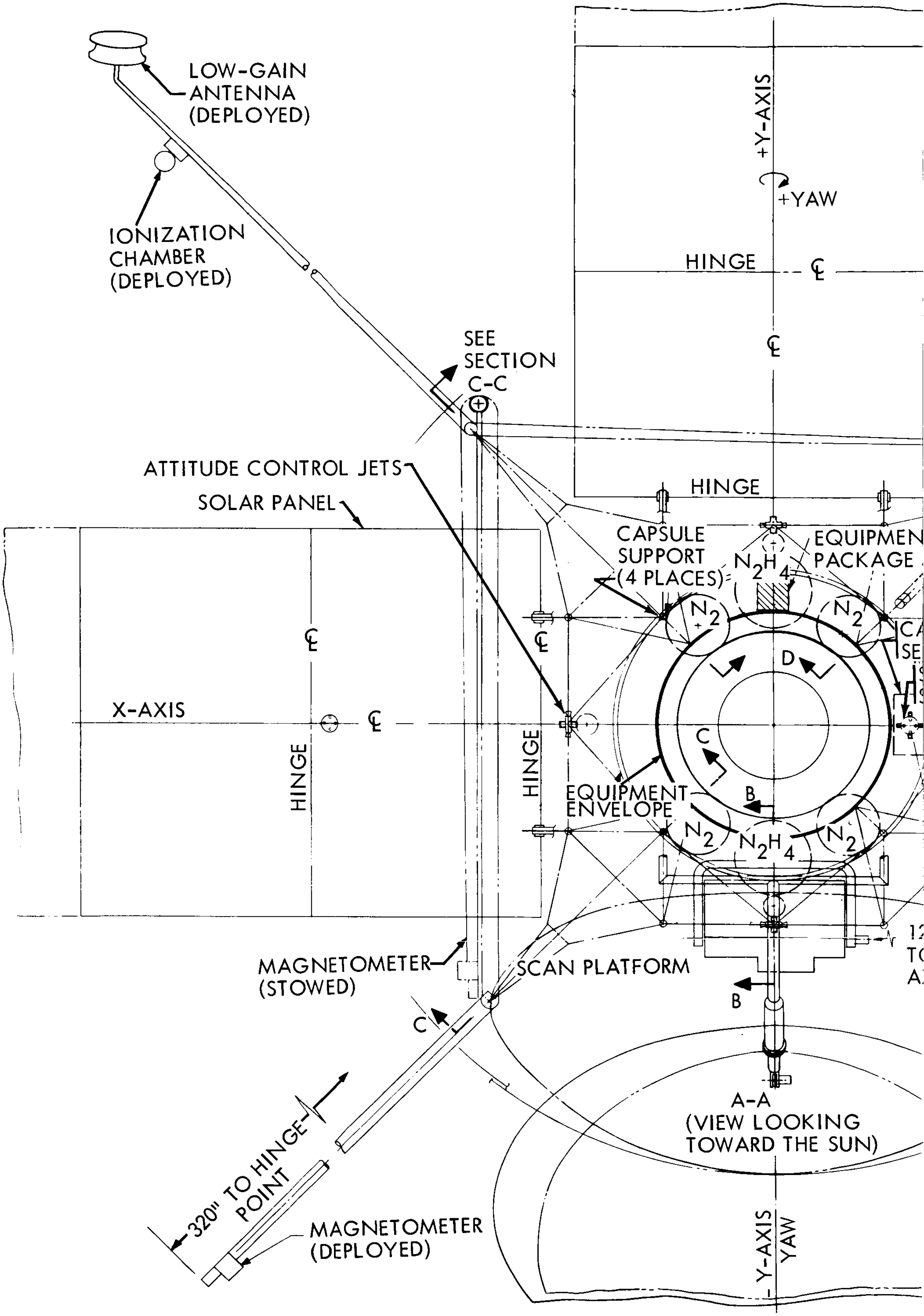
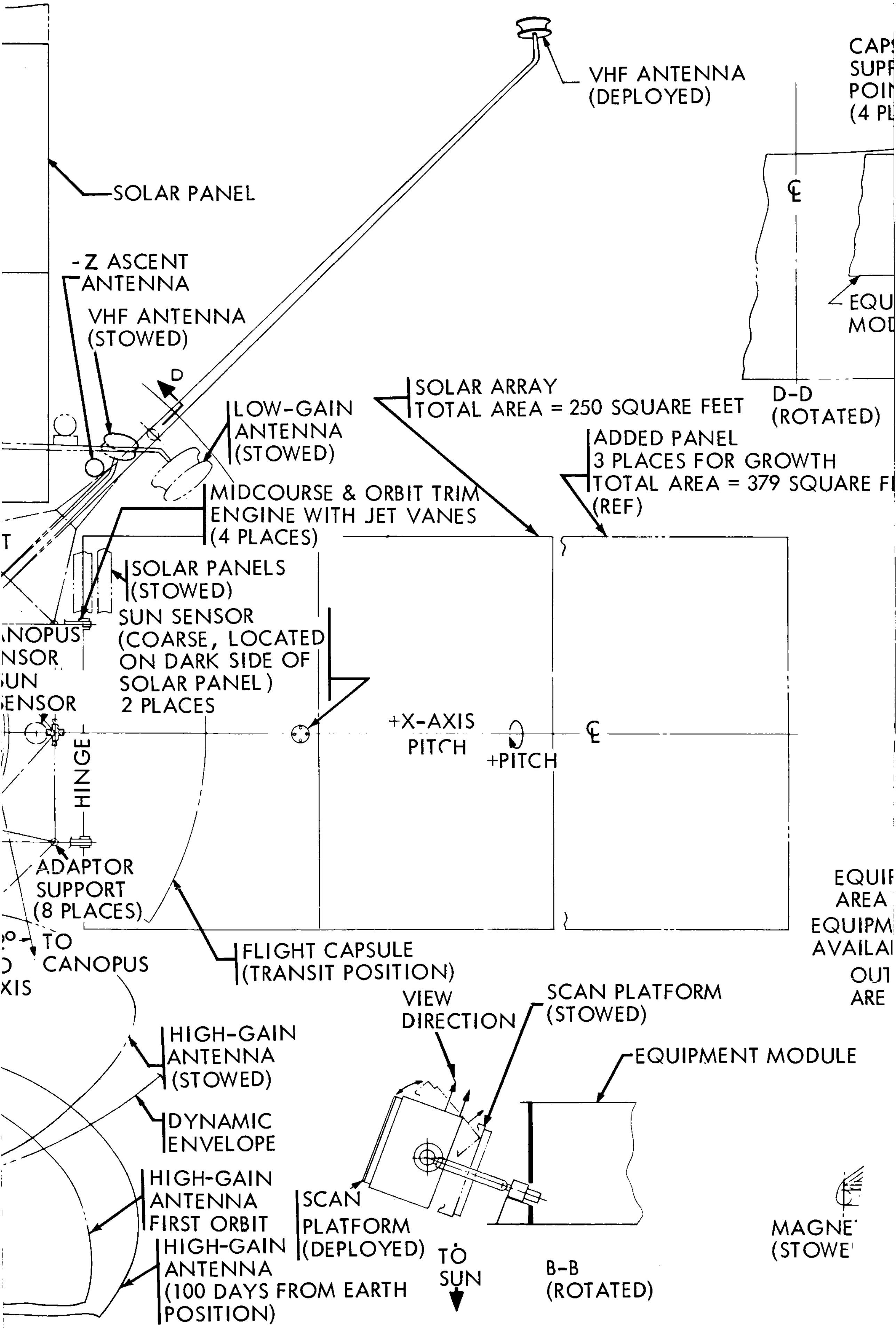


Figure 3.10-1: Flight Spacecraft Configuration



LOW-GAIN ANTENNA (DEPLOYED)
IONIZATION CHAMBER (DEPLOYED)
+Y-AXIS
+YAW
HINGE
SEE SECTION C-C
ATTITUDE CONTROL JETS
SOLAR PANEL
HINGE
CAPSULE SUPPORT (4 PLACES)
N_2H_4
EQUIPMEN
PACKAGE
N_2
N_2
D
X-AXIS
HINGE
HINGE
C
EQUIPMENT ENVELOPE
B
N_2
N_2H_4
N_2
MAGNETOMETER (STOWED)
SCAN PLATFORM
B
C
320" TO HINGE POINT
MAGNETOMETER (DEPLOYED)
A-A
(VIEW LOOKING TOWARD THE SUN)
-Y-AXIS
YAW

243 (1)


VHF ANTENNA (DEPLOYED)
SOLAR PANEL
-Z ASCENT ANTENNA
VHF ANTENNA (STOWED)
D
LOW-GAIN ANTENNA (STOWED)
SOLAR ARRAY TOTAL AREA = 250 SQUARE FEET
D-D (ROTATED)
ADDED PANEL 3 PLACES FOR GROWTH TOTAL AREA = 379 SQUARE F
(REF)
MIDCOURSE & ORBIT TRIM ENGINE WITH JET VANES (4 PLACES)
SOLAR PANELS (STOWED)
SUN SENSOR (COARSE, LOCATED ON DARK SIDE OF SOLAR PANEL) 2 PLACES
NOPUS NSOR UN ENSOR
+X-AXIS PITCH
+PITCH
HINGE
ADAPTOR SUPPORT (8 PLACES)
TO CANOPUS
XIS
FLIGHT CAPSULE (TRANSIT POSITION)
VIEW DIRECTION
SCAN PLATFORM (STOWED)
EQUIPMENT MODULE
HIGH-GAIN ANTENNA (STOWED)
DYNAMIC ENVELOPE
HIGH-GAIN ANTENNA FIRST ORBIT
HIGH-GAIN ANTENNA (100 DAYS FROM EARTH POSITION)
SCAN PLATFORM (DEPLOYED)
TO SUN
B-B (ROTATED)
CAP SUPP POIN (4 PL
EQU MOD
EQUIP AREA EQUIPM AVAILA OUT ARE
MAGNE (STOWE

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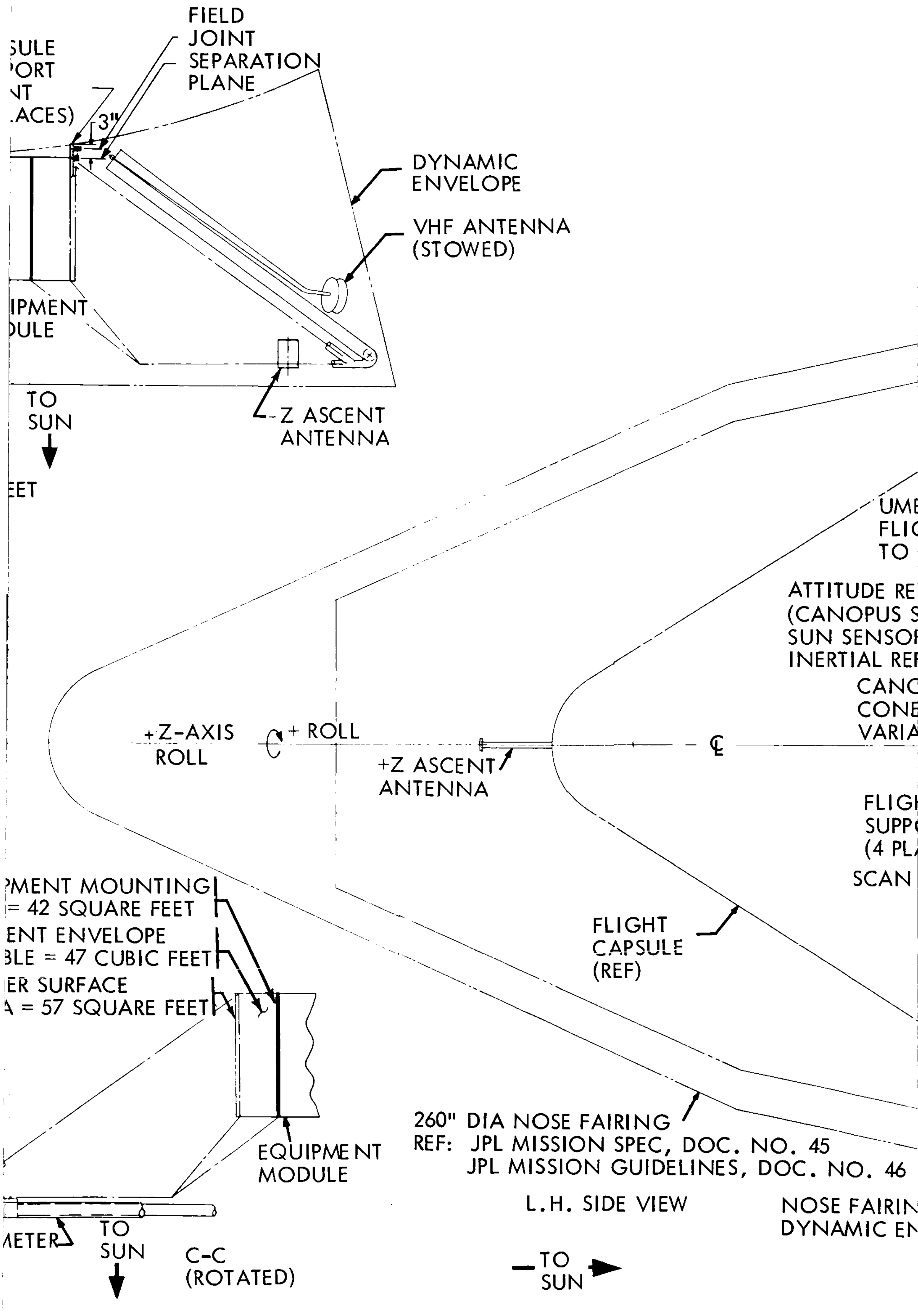
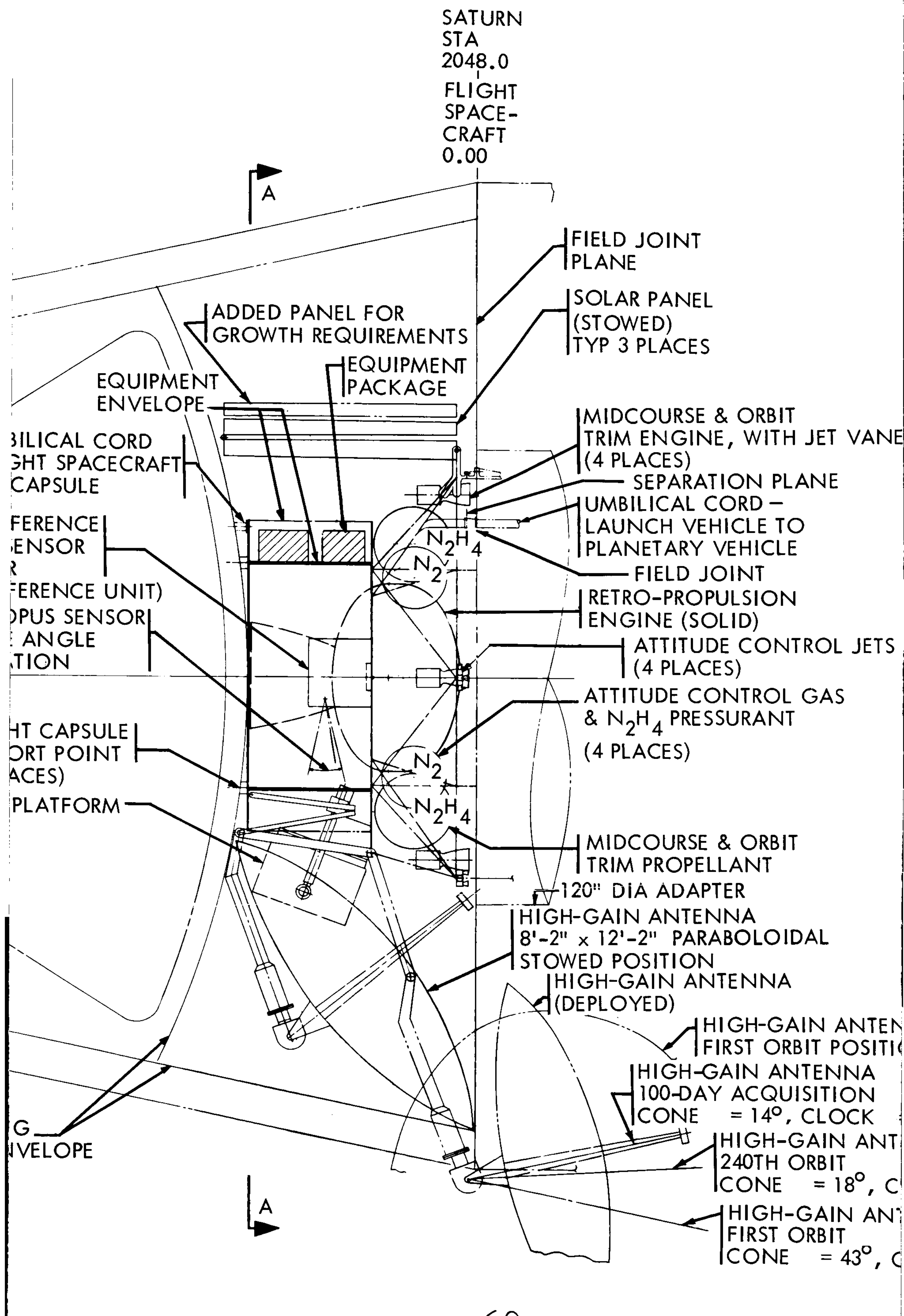


FIELD JOINT SEPARATION PLANE
3"
DYNAMIC ENVELOPE
VHF ANTENNA (STOWED)
-Z ASCENT ANTENNA
TO SUN
+Z-AXIS ROLL
+ ROLL
+Z ASCENT ANTENNA
℄
ATTITUDE RE
(CANOPUS S
SUN SENSOR
INERTIAL REF
FLIGHT CAPSULE (REF)
= 42 SQUARE FEET
= 47 CUBIC FEET
= 57 SQUARE FEET
EQUIPMENT MODULE
TO SUN
C-C (ROTATED)
260" DIA NOSE FAIRING
REF: JPL MISSION SPEC, DOC. NO. 45
JPL MISSION GUIDELINES, DOC. NO. 46
L.H. SIDE VIEW
TO SUN


SATURN
STA
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FLIGHT
SPACE-
CRAFT
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A
FIELD JOINT
PLANE
SOLAR PANEL
(STOWED)
TYP 3 PLACES
ADDED PANEL FOR
GROWTH REQUIREMENTS
EQUIPMENT
PACKAGE
EQUIPMENT
ENVELOPE
BILICAL CORD
GHT SPACECRAFT
CAPSULE
FERENCE
SENSOR
R
FERENCE UNIT)
OPUS SENSOR
E ANGLE
ATION
MIDCOURSE & ORBIT
TRIM ENGINE, WITH JET VANE
(4 PLACES)
SEPARATION PLANE
UMBILICAL CORD –
LAUNCH VEHICLE TO
PLANETARY VEHICLE
FIELD JOINT
N_2H_4
N_2
RETRO-PROPULSION
ENGINE (SOLID)
ATTITUDE CONTROL JETS
(4 PLACES)
ATTITUDE CONTROL GAS
& N_2H_4 PRESSURANT
(4 PLACES)
HT CAPSULE
ORT POINT
ACES)
PLATFORM
N_2
N_2H_4
MIDCOURSE & ORBIT
TRIM PROPELLANT
120" DIA ADAPTER
HIGH-GAIN ANTENNA
8'-2" x 12'-2" PARABOLOIDAL
STOWED POSITION
HIGH-GAIN ANTENNA
(DEPLOYED)
HIGH-GAIN ANTEN
FIRST ORBIT POSITI
HIGH-GAIN ANTENNA
100-DAY ACQUISITION
CONE = 14°, CLOCK
G
NVELOPE
HIGH-GAIN ANT
240TH ORBIT
CONE = 18°, C
A
HIGH-GAIN ANT
FIRST ORBIT
CONE = 43°, C

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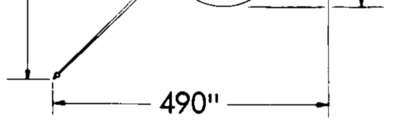


SPACECRAFT DEPLOYED
(VIEW LOOKING TOWARD SUN)

INA
ƆN

= 210°
ENNA

LOCK = 265°
ENNA

LOCK = 283°

Figure 3.10-2: Model 945-6026 Mars Exploration Flight Spacecraft

5) A separately mounted ultra-violet spectrometer oriented to view the planet atmosphere over approximately one-quarter of the circumference.

6) An 8 ft. 2-in. by 12 ft. 2-in. paraboloidal antenna.

7) A boom for mounting the magnetometer. This boom is shown extending 320 inches from the hinge, although the final length will be determined as the spacecraft magnetic field becomes defined.

8) Separately mounted omni-, VHF-, and two ascent-antennas.

9) Canopus tracker, fine Sun-sensor and IRU mounted in a single package for dimensional control. The package is located on reference plane B and the X-axis.

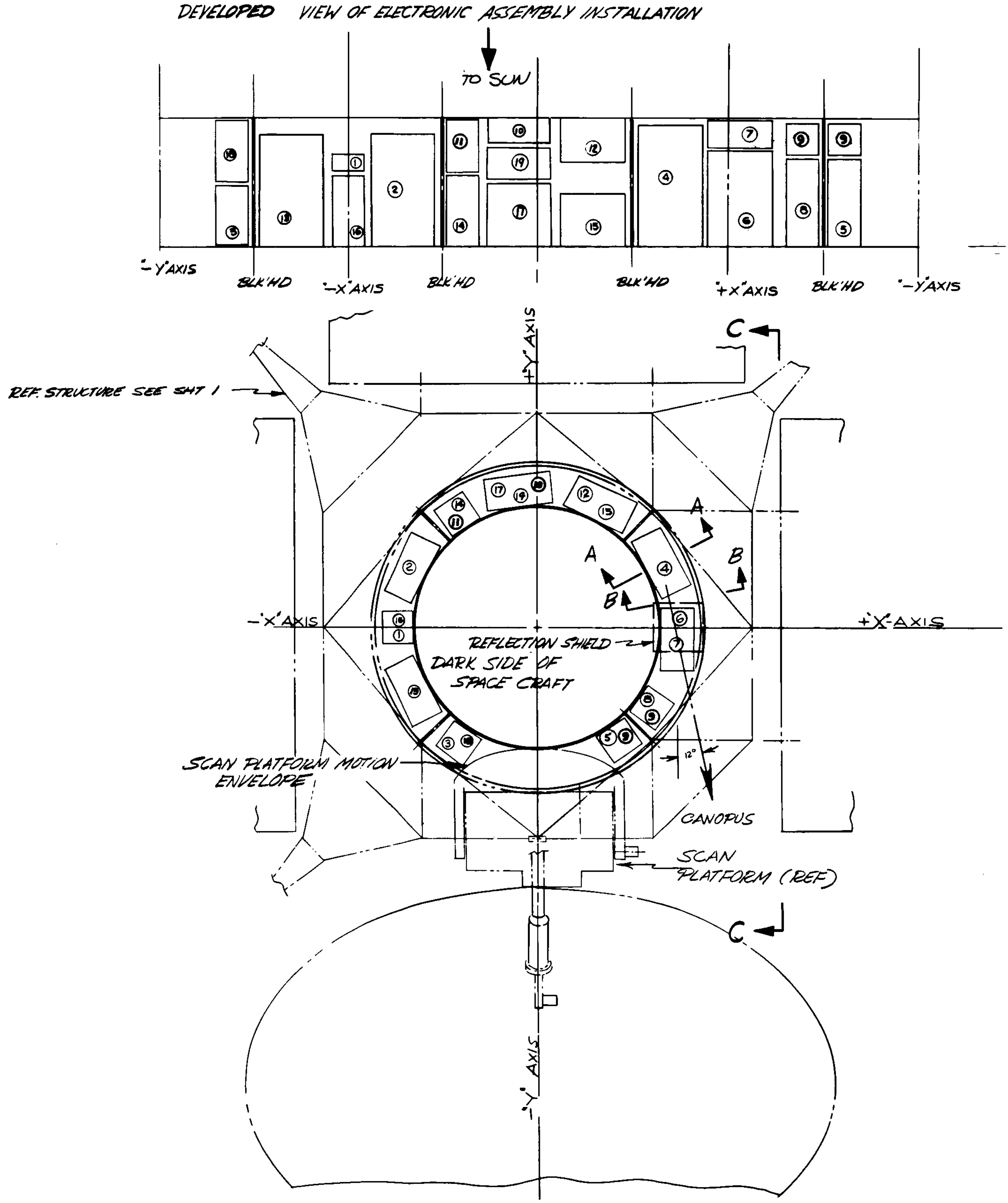
10) A modular midcourse- and orbit-insertion-propulsion subsystem. This subsystem is assembled as a unit for orbiting configurations. For flyby configurations, the midcourse propulsion sybsystem components are rearranged slightly but no new or changed components are required.

11) A cold-gas (N_2) reaction-control subsystem. This subsystem is assembled as a part of the propulsion subsystem module.

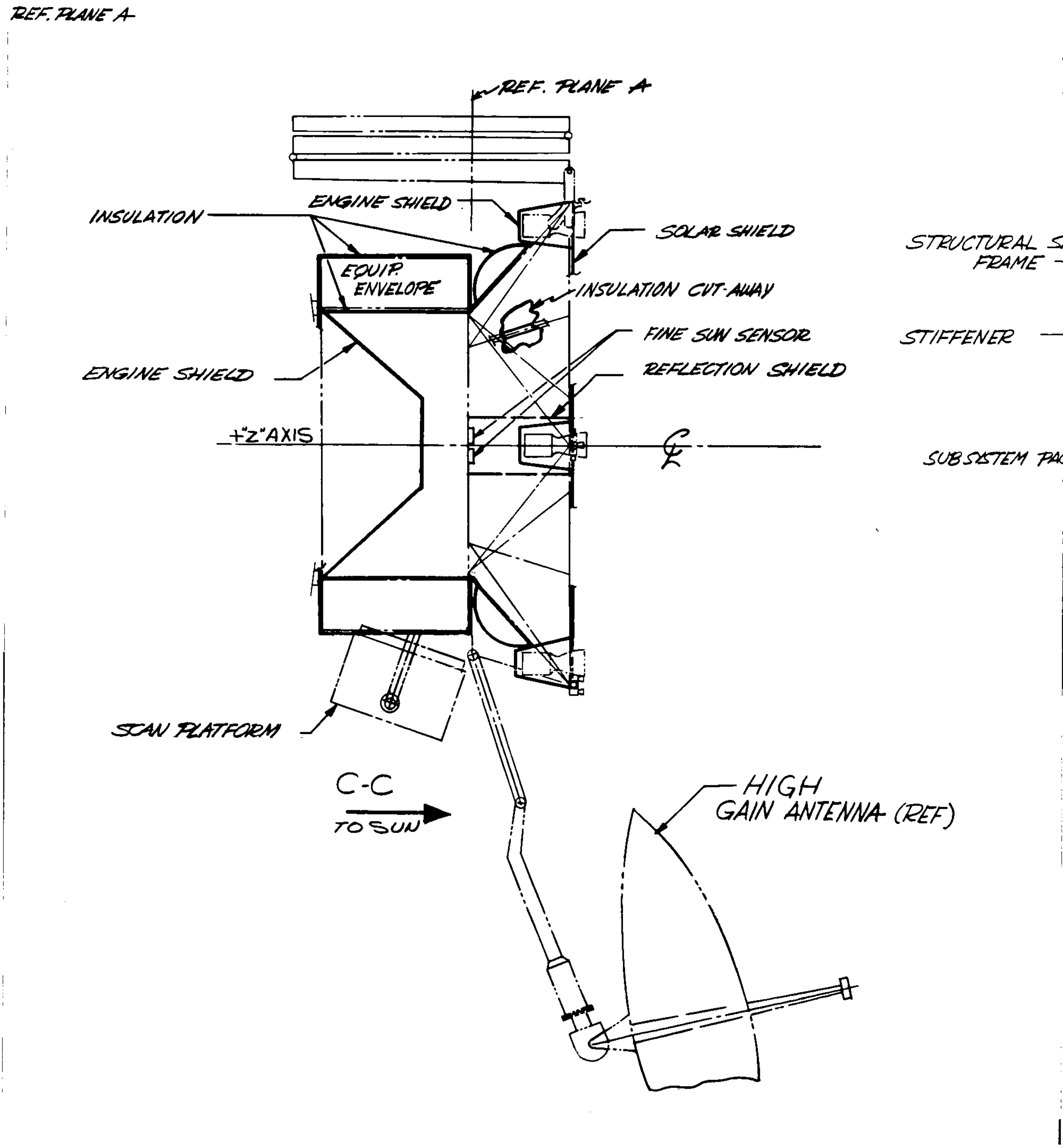
12) Electronic packages are mounted on the exterior surface of the cylindrical upper structure (equipment-support module). This mounting arrangement provides easy access to electrical connectors through removal of nonstructural insulation panels.

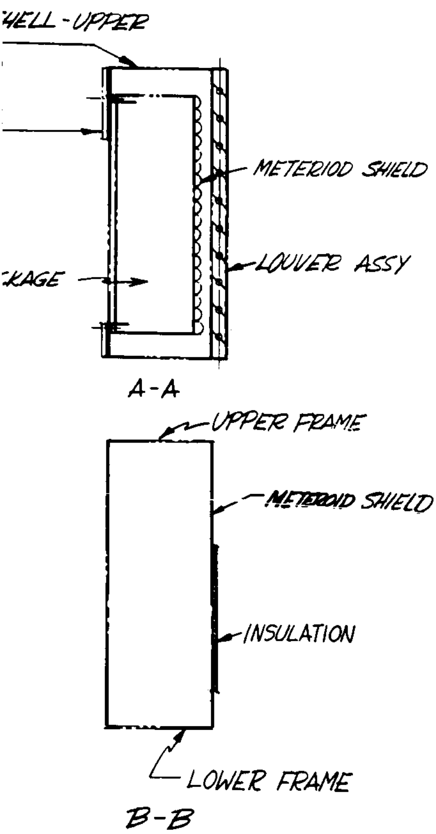
The arrangement of the electronic packages on the spacecraft are shown in Figure 3.10-3. The installation of the packages is defined in Section 4.4.2.

A functional block diagram is shown in Figure 3.10-4.


DEVELOPED VIEW OF ELECTRONIC ASSEMBLY INSTALLATION
TO SUN
-Y AXIS
BLK'HD
-X AXIS
BLK'HD
BLK'HD
+X AXIS
BLK'HD
-Y AXIS
+Y AXIS
C
REF. STRUCTURE SEE SHT 1
A
A
B
B
-X AXIS
+X AXIS
REFLECTION SHIELD
DARK SIDE OF
SPACE CRAFT
12°
SCAN PLATFORM MOTION
ENVELOPE
CANOPUS
SCAN
PLATFORM (REF)
C
-Y AXIS

2470

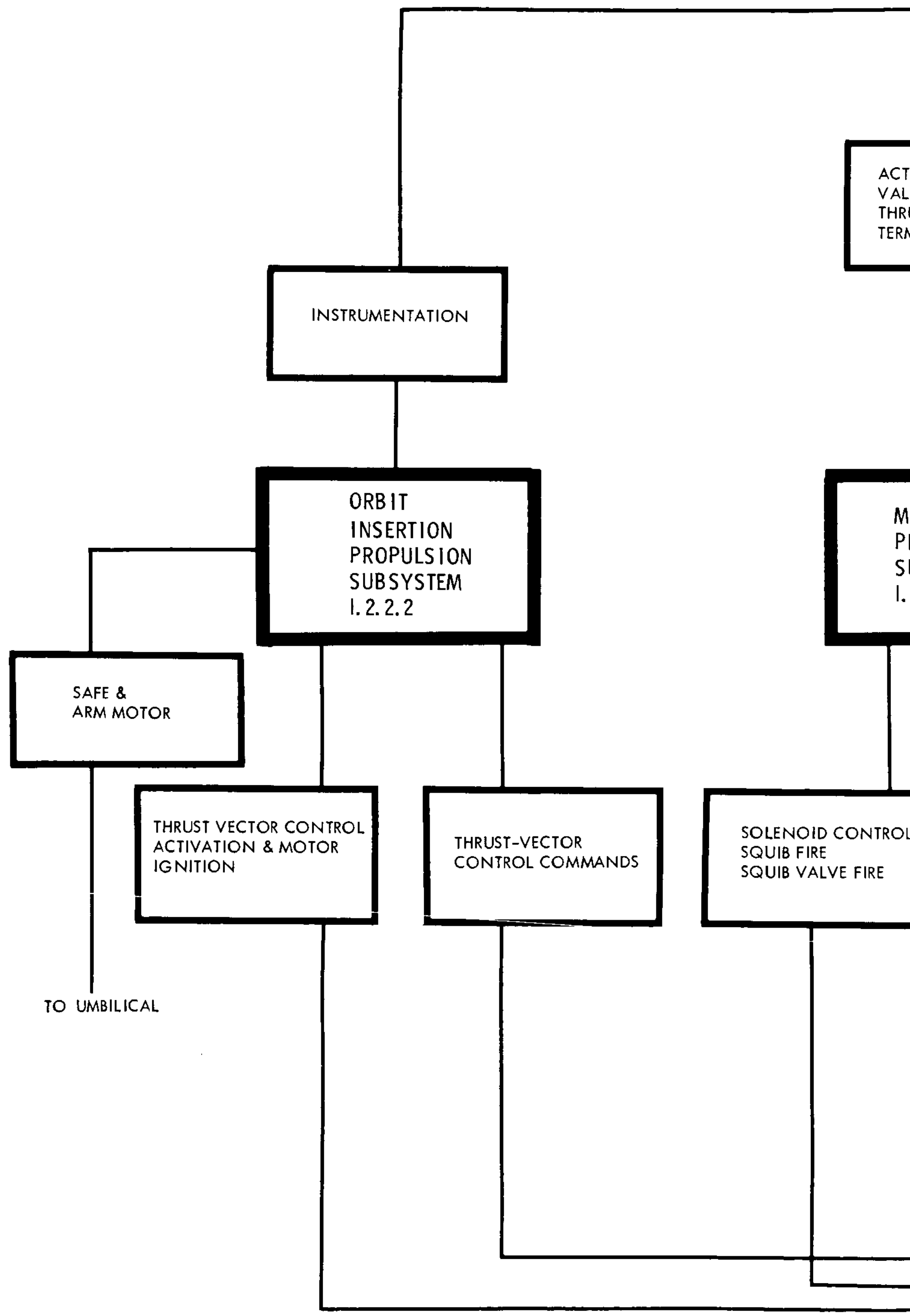

REF. PLANE A
REF. PLANE A
INSULATION
ENGINE SHIELD
SOLAR SHIELD
EQUIP. ENVELOPE
INSULATION CUT-AWAY
FINE SUN SENSOR
REFLECTION SHIELD
ENGINE SHIELD
+"Z" AXIS
SCAN PLATFORM
C-C
TO SUN
HIGH GAIN ANTENNA (REF)
STIFFENER
FRAME



BOX	ITEM	SUBSYSTEMS
1		RELAY RADIO SUBSYSTEM
	1	DETECTOR & BIT SYSNC.
	2	RECEIVER
2		TELEMETRY & DATA STORAGE
	1	PLANETARY SCIENCE TAPE RECORDER
	2	TAPE RECORDER #1
	3	" " #2
3		TELEMETRY PROCESSOR
	1	ENG. CORE MEMORY
	2	CAPSULE CORE MEMORY
	3	MULTIPLEXER-ENCODER & MODULATOR
	4	POWER SUPPLY
4		S/C RADIO
	1	R.F. POWER AMPLIFIER UNIT
	2	POWER SUPPLY
	3	CIRCULATOR SWITCH
	4	PREAMPLIFIER
	5	DIPLEXER
	6	PRESELECTOR
	7	INSULATOR
	8	TWT AMPLIFIER #1 & #2
	9	INSULATOR (3)
	10	BAND PASS FILTER (3)
	11	BAND STOP
	12	HYBRID
	13	CIRCULATOR SWITCH
	14	STRUCTURE
5		RECEIVERS & EXCITORS (2)
	1	RECEIVERS (2)
	2	COMMAND DECTORS (2)
	3	RANGING UNIT (2)
	4	EXCITOR & LAUNCH TRANSMITTER
	5	LAUNCH TRANSMITTER
	6	REDUNDANCY CONTROL & LOGIC
6		ATTITUDE REFERENCE
	1	IRU
	2	CANOPUS
	3	SUN SENSOR
7		AUTO PILOT
	1	SCYRONIZER
		CENTRAL COMPUTER & SEQUENCER
8	1	CONTROL ASSY
9	2	SWITCHING ASSY
10		BATTERY
11		"
12	1	BATTERY CHARGER BOOSTER ASSY
	2	P.S. & L
	3	BATTERY
	4	SENSOR
13	1	DC/DC REGULATOR
	2	FAILURE SENSOR
14	1	DC/AC 2400 INVERTER
	2	FAILURE SENSOR DC/AC
	3	DC/AC 400 ~ 1Ø
15	1	IR SPECTROMETER ELECT
	2	RADIO BEACON
	3	R.F. NOISE DECTOR
16	1	MAGNETOMETER ELECTRONICS
	2	T.V. CAMERA ELECTRONICS
	3	IONIZATION CHAMBER ELECTRONICS
17		U.V. SPECTROMETER
18		D.A.E
19		D.A.E

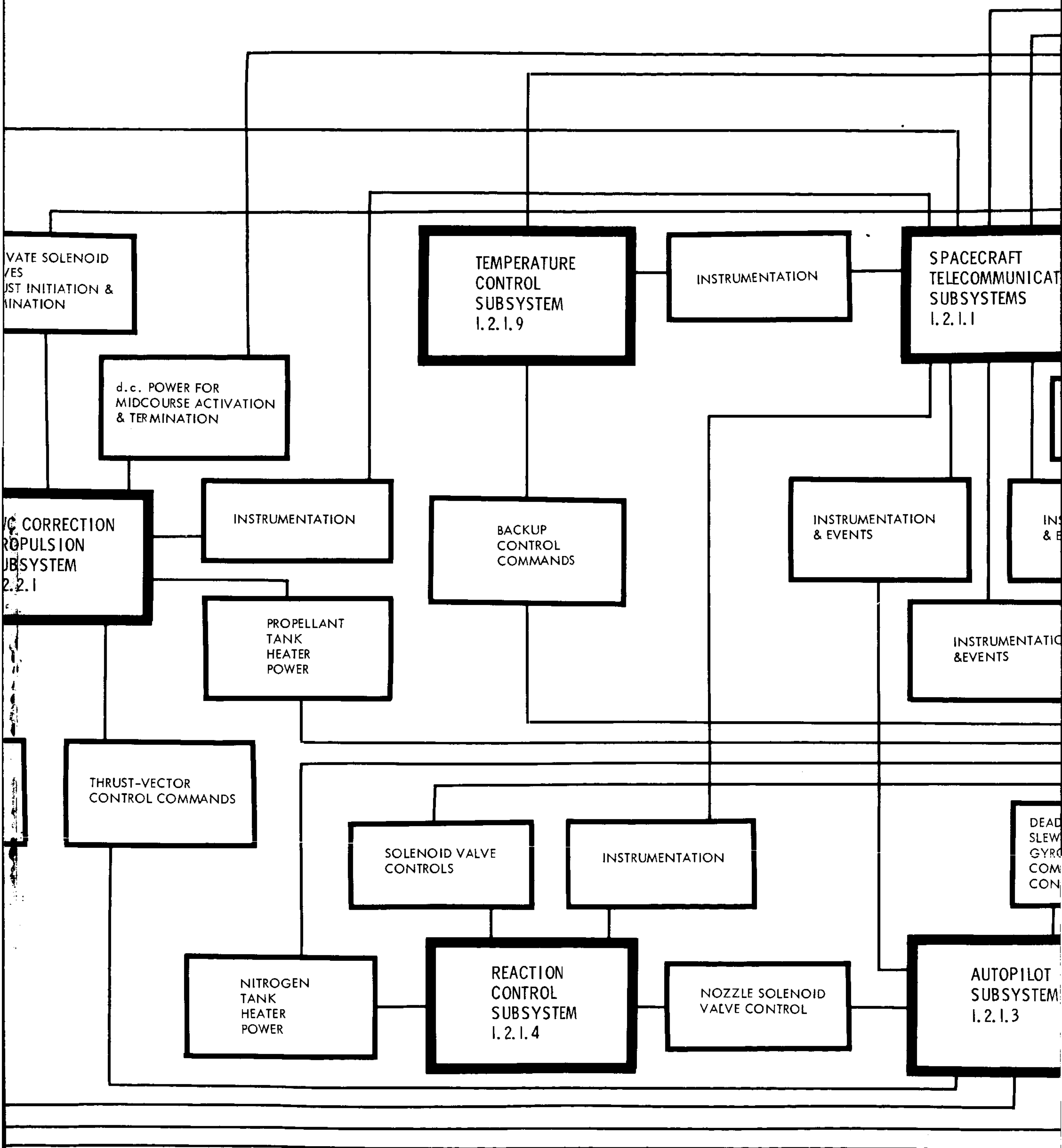
Figure 3.10-3: Inboard Profile — Flight Spacecraft

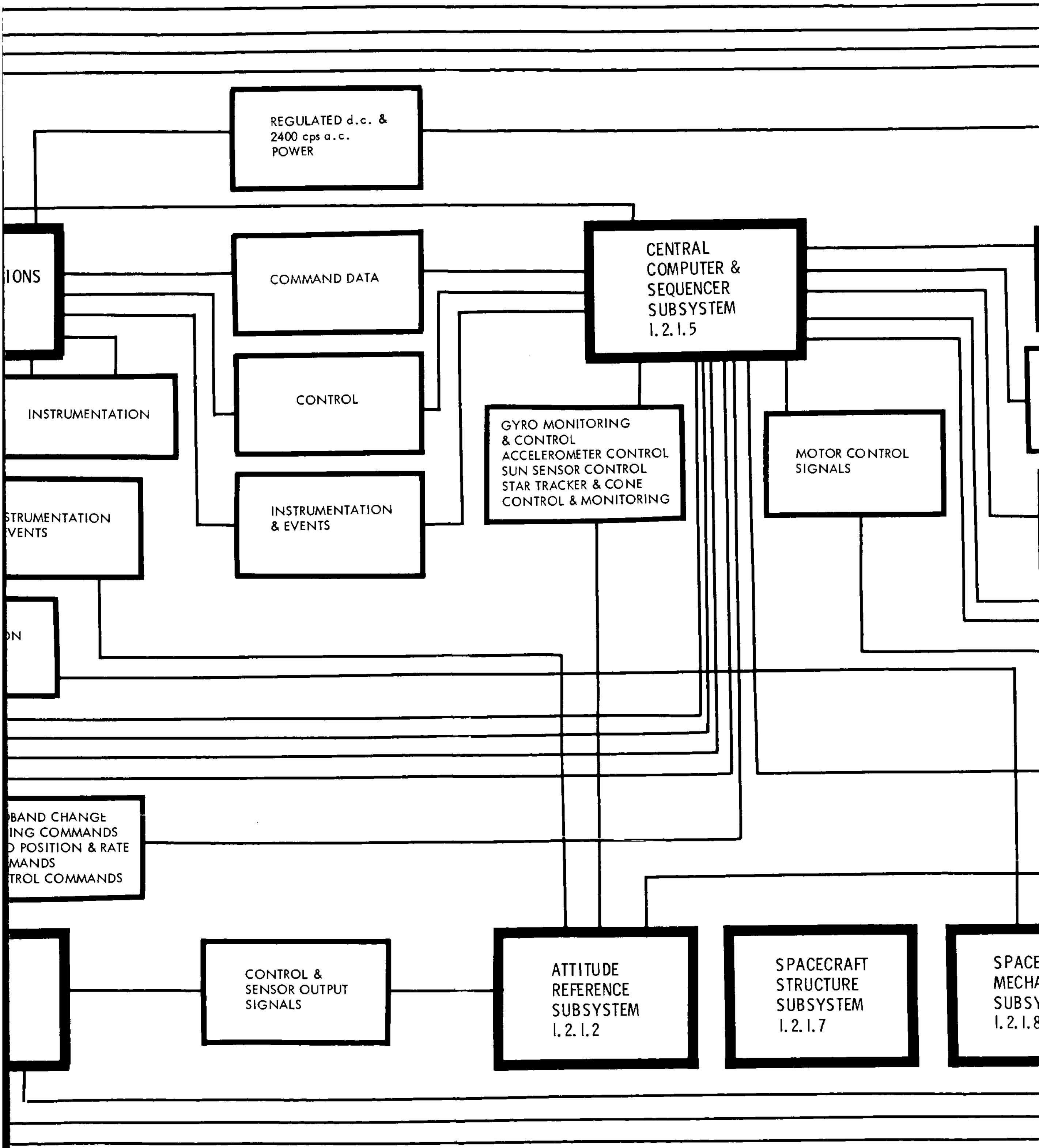
3



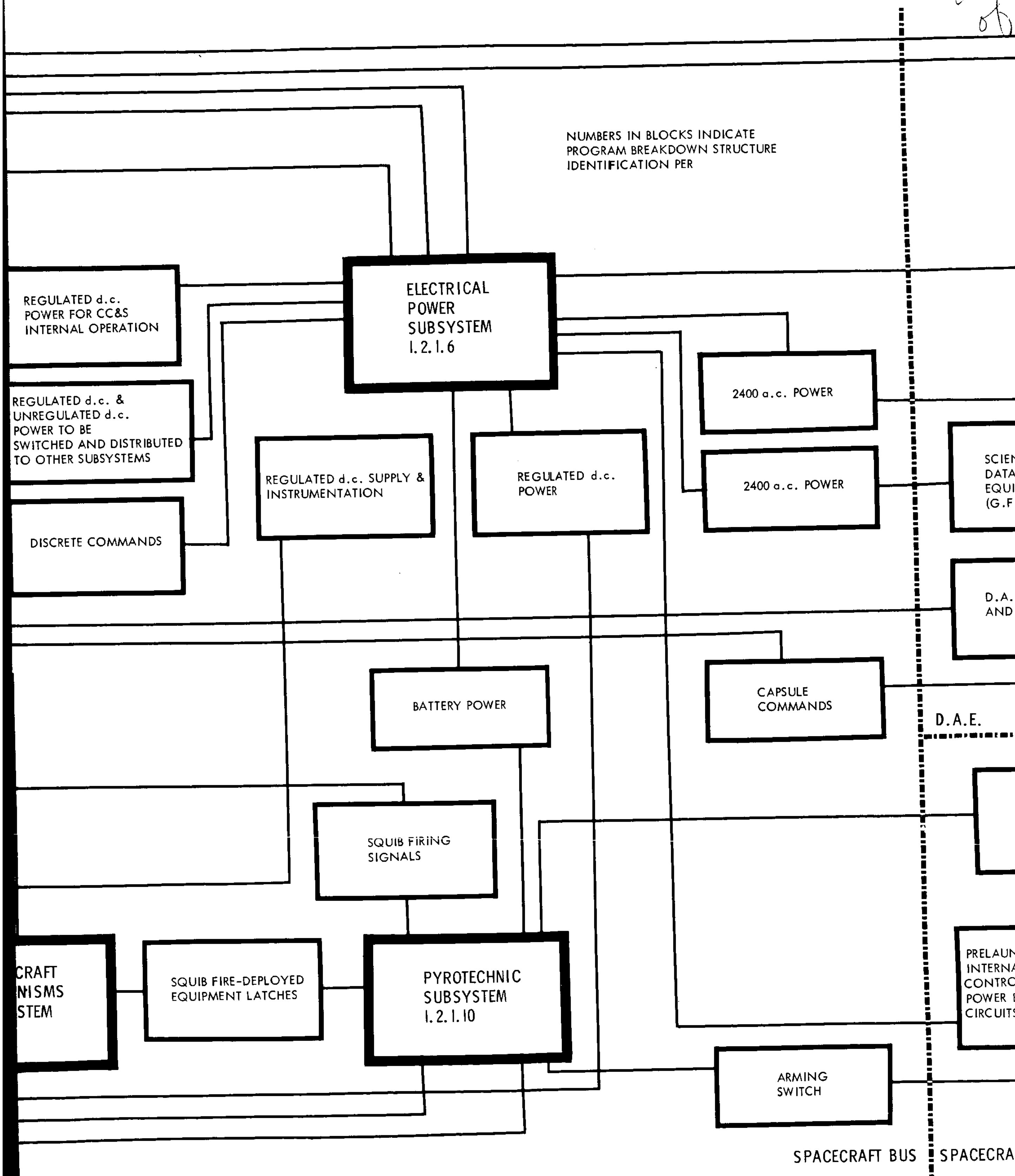
INSTRUMENTATION
ORBIT INSERTION PROPULSION SUBSYSTEM 1.2.2.2
SAFE & ARM MOTOR
TO UMBILICAL
THRUST VECTOR CONTROL ACTIVATION & MOTOR IGNITION
THRUST-VECTOR CONTROL COMMANDS
SOLENOID CONTROL
SQUIB FIRE
SQUIB VALVE FIRE

249 (1)


VATE SOLENOID
/ES
JST INITIATION &
MINATION
d.c. POWER FOR MIDCOURSE ACTIVATION & TERMINATION
TEMPERATURE CONTROL SUBSYSTEM 1.2.1.9
INSTRUMENTATION
SPACECRAFT TELECOMMUNICAT
SUBSYSTEMS 1.2.1.1
C CORRECTION
ROPULSION
JBSYSTEM
2.2.1
INSTRUMENTATION
BACKUP CONTROL COMMANDS
INSTRUMENTATION & EVENTS
INS
& E
PROPELLANT TANK HEATER POWER
INSTRUMENTATIO
&EVENTS
THRUST-VECTOR CONTROL COMMANDS
SOLENOID VALVE CONTROLS
INSTRUMENTATION
DEAD
SLEW
GYRO
COM
CON
NITROGEN TANK HEATER POWER
REACTION CONTROL SUBSYSTEM 1.2.1.4
NOZZLE SOLENOID VALVE CONTROL
AUTOPILOT SUBSYSTEM 1.2.1.3


REGULATED d.c. & 2400 cps a.c. POWER
IONS
COMMAND DATA
CENTRAL COMPUTER & SEQUENCER SUBSYSTEM 1.2.1.5
INSTRUMENTATION
CONTROL
STRUMENTATION VENTS
INSTRUMENTATION & EVENTS
GYRO MONITORING & CONTROL ACCELEROMETER CONTROL SUN SENSOR CONTROL STAR TRACKER & CONE CONTROL & MONITORING
MOTOR CONTROL SIGNALS
N
BAND CHANGE
ING COMMANDS
POSITION & RATE
MANDS
TROL COMMANDS
CONTROL & SENSOR OUTPUT SIGNALS
ATTITUDE REFERENCE SUBSYSTEM 1.2.1.2
SPACECRAFT STRUCTURE SUBSYSTEM 1.2.1.7
SPACE
MECHA
SUBSY
1.2.1.8

249 (3)


NUMBERS IN BLOCKS INDICATE
PROGRAM BREAKDOWN STRUCTURE
IDENTIFICATION PER
ELECTRICAL POWER SUBSYSTEM 1.2.1.6
REGULATED d.c. POWER FOR CC&S INTERNAL OPERATION
REGULATED d.c. & UNREGULATED d.c. POWER TO BE SWITCHED AND DISTRIBUTED TO OTHER SUBSYSTEMS
DISCRETE COMMANDS
REGULATED d.c. SUPPLY & INSTRUMENTATION
REGULATED d.c. POWER
2400 a.c. POWER
2400 a.c. POWER
BATTERY POWER
CAPSULE COMMANDS
D.A.E.
SQUIB FIRING SIGNALS
CRAFT
NISMS
STEM
SQUIB FIRE-DEPLOYED EQUIPMENT LATCHES
PYROTECHNIC SUBSYSTEM 1.2.1.10
ARMING SWITCH
SPACECRAFT BUS
SPACECRA

e does
u go out?
DAE

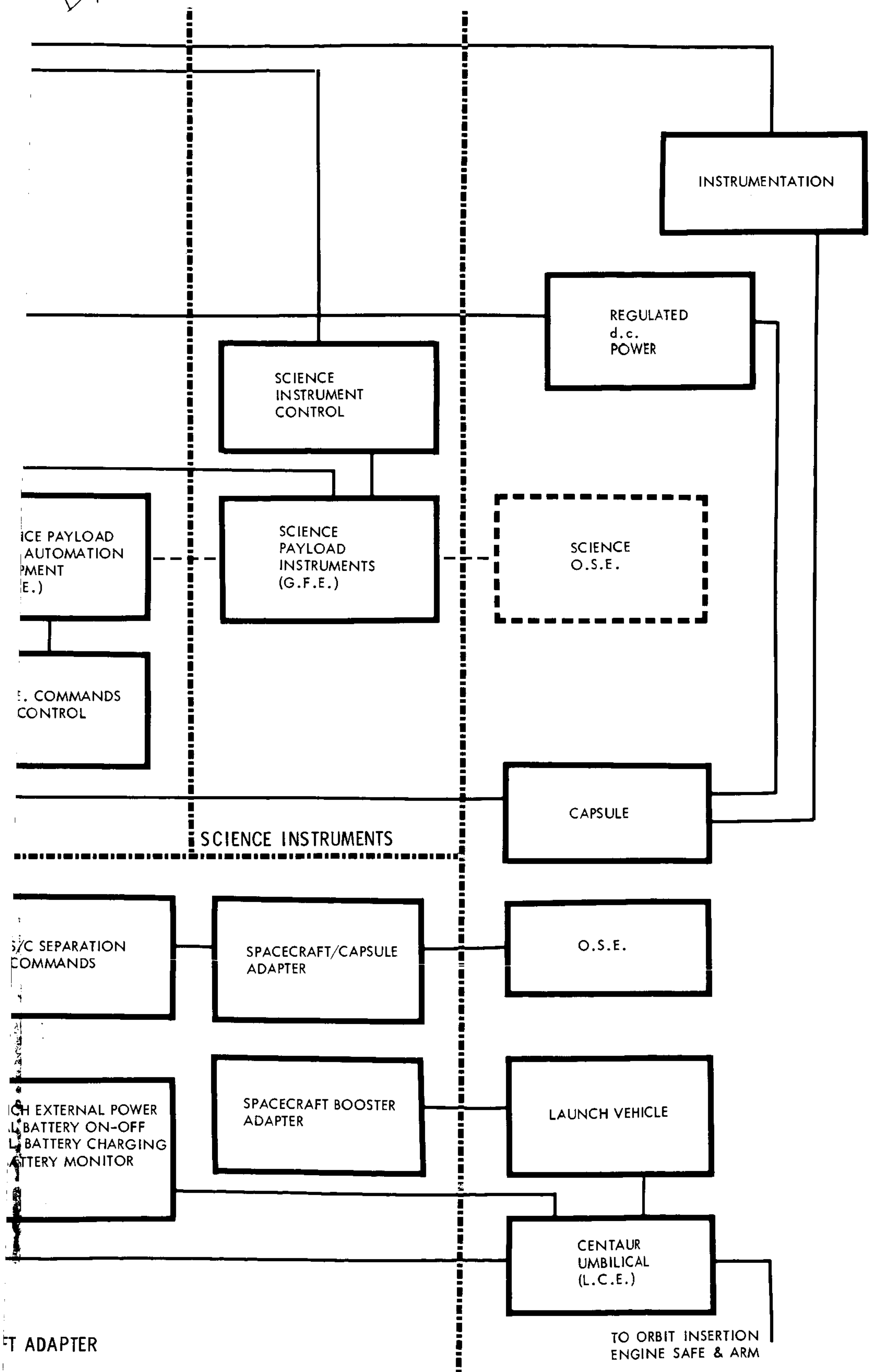


Figure 3.10-4: Functional Block Diagram Flight Spacecraft

6

3.10.3.2 Detailed Structural Alignments for Voyager

The tolerances described in the following sections are those determined to be required for proper operation of the spacecraft, and are those that have been used in all preliminary design calculations. As the spacecraft design progresses and more detailed data becomes available, these tolerances will be reviewed and changed as required.

It is intended that an alignment fixture be constructed and used throughout the fabrication and assembly of all spacecraft to ensure proper alignment maintenance of all critical elements. This fixture will be designed and developed together with the quality control department so that a common basic fixture can be used for manufacture and quality control checking. The fixture will be such that the spacecraft can be returned to it at any time for critical alignment adjustments and prelaunch checks whenever desired. Consideration will be given to a requirement for a duplicate fixture at the launch site, and a similar or less elaborate fixture can be used to check the effects of space environment on alignments in a test chamber if desired for the proof test model.

The design requirements for the spacecraft will include provisions for maintaining alignments during shipping and handling, boost accelerations and vibrations, and thermal warpage during the extreme temperature environments of space.

The alignments on the assembled spacecraft will be checked and recorded for each spacecraft. Checking will be done in a simply supported, vertical position with sufficient auxilliary support to simulate zero-gravity conditions on overhanging members.

Spacecraft Reference Axes and Planes--Coordinate System--The spacecraft coordinate system consists of three mutually perpendicular reference axes as shown in Figure 3.10-5. The X and Y axes lie in a plane (plane A) parallel to the spacecraft-launch vehicle interface plane. The Z axis is perpendicular to this plane and is located at the centerline of the spacecraft to coincide with the launch vehicle centerline. The pitch-reference axis is the X axis and positive pitch is clockwise when the spacecraft is viewed from the side opposite the side containing the attitude reference subsystem. The yaw axis reference is the Y axis and positive yaw is clockwise when the spacecraft is viewed from the high-gain-antenna side of the spacecraft. The Z axis is the reference for roll, and positive direction is defined as clockwise when the spacecraft is viewed from the Sun side.

Plus-X is measured away from the spacecraft centerline (Z axis) in the direction toward the attitude reference subsystem. Minus-X is measured away from the spacecraft in the direction opposite the attitude references.

Plus-Y is measured in the direction generally away from Canopus and minus-Y is in the direction generally toward Canopus.

Plus-Z is measured in a direction away from the Sun from reference-plane A. Minus-Z is measured in a direction toward the Sun from

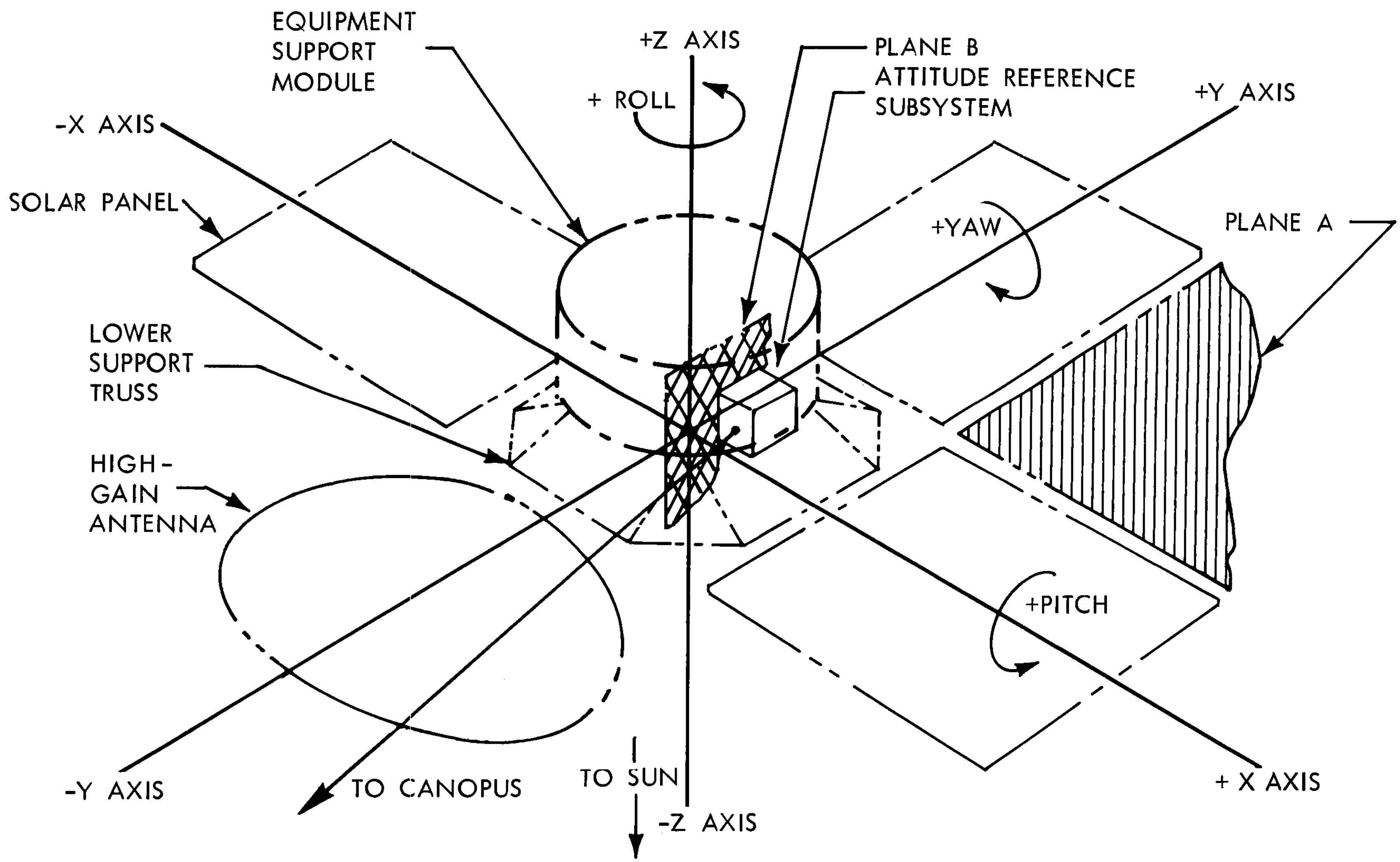


Figure 3.10-5: Boeing Voyager Coordinate System

reference plane A, and, therefore, extends down from the spacecraft.

The plane in which the X and Y axes lie is defined as the base plane or plane A. It is defined as spacecraft station 0 (Saturn Station 2048.0) and spacecraft stations are numbered positively up (in a plus-Z direction), from this base. It is located at the spacecraft launch vehicle interface.

A secondary reference plane B is established as being normal to reference plane A and passing through the attitude reference subsystem mounting plate that is located on the plus-X side of the spacecraft. This plane shall be perpendicular to the reference plane A within 0.20 milliradians.

Attitude Reference Subsystem Alignment--The attitude reference subsystem package is provided with mounting guide pins that must align with reference plane B exactly, and be parallel to reference plane A with ±0.5 milliradians. The vertical position of these mounting guide pins must be controlled to the reference plane A within ±0.025 inches.

Canopus-Sensor Mounting--The Canopus sensor is located within the attitude reference subsystem package. It must be aligned to the package guide pins and mounting base within ±0.5 milliradian and shall be oriented 12 degrees from reference plane B and in a plane parallel to reference plane A. Alignment prisms will be used to align the sensor.

Fine Sun-Sensor Mounting--The fine Sun-sensor is located in the attitude reference subsystem package. It must be aligned to the package mounting base and the guide pins within ±0.5 milliradians.

Inertial Reference Unit--The IRU is mounted in the attitude reference subsystem package. It must be aligned to the package mounting base and guide pins within ±0.5 milliradian.

Coarse Sun-Sensor Mounting--Coarse Sun-sensors are mounted on the dark side of the solar panels along the X axis. These sensors must align with reference plane A within ±0.5 degree.

Additional coarse Sun sensors are mounted on the base of the spacecraft (on the Sun side). They are mounted directly to the structure in reference plane A.

Sun Gates Mounting--Sun gates are mounted parallel and perpendicular to reference plane A within ± 10 minutes.

Magnetometer Mounting--The magnetometer is mounted on an extendable boom in the -Y, -X quadrant. It will be aligned with the Z axis within ±1 degree in all directions.

High-Gain-Antenna Mounting--The high-gain antenna is mounted on a deployable boom and driven by a servomechanism. The extended and locked position of the antenna, with the drive system in its neutral position, will be aligned to the theoretically required angle within ±2 milliradians.

Propulsion and Reaction Control Subsystem Alignment--The propulsion and reaction control subsystem for the 1971 flight is assembled as a modular unit containing the midcourse engines, the orbit insertion engine reaction control nozzles, and all tankage and plumbing. This module will be prealigned as follows:

1) The thrust axis of each midcourse engine will be aligned to the orbit insertion engine thrust axis within ±0.25 degree.
2) The orbit insertion engine will be located so that its thrust axis passes through the center of gravity of the module within 0.03 inch. The module center of gravity will be determined experimentally on each module.
3) The reaction-control nozzles will be located with the thrust axes normal to reference plane A within ±2 degrees.

The module will be installed in the spacecraft so that thrust centerline of the orbit insertion engine lies within a circle 0.06 inch in diameter in a plane parallel to reference plane A and passing through the spacecraft center of gravity. The center of gravity in this plane will be determined experimentally on each spacecraft. The station location of the center of gravity will be calculated.

Science Scan Platform Mounting--The science-scan platform is mounted on the -Y side of the spacecraft with a two-axis gimbal.

The axis in the X direction will be aligned parallel to reference plane A and normal to reference plane B within ± 0.5 milliradian.

Omni-antenna Mounting--The omni-antenna, located on a boom in the -X, +Y quadrant will be aligned so that the antenna axis is at an angle of 175 degrees to reference plane A and 87 degrees to reference plane B measured clockwise, and when viewed in the direction of the -X and -Z axis respectively. These angles will be held within ± 1 degree.

VHF Antenna Mounting--The VHF antenna is located on a boom in the +X, +Y quadrant. Its antenna axis will be aligned so that it has an angle of 190 degrees to reference plane A and 175 degrees to reference plane B measured clockwise, and when viewed in the direction of the -X and -Z axis respectively. These angles will be held within ± 1 degree.

Science-Instrument Alignment--The science payload is not currently defined. Alignment requirements will be added when specific instruments are known.

Solar Panels--Solar panels will be parallel to reference plane A within ± 0.5 degree.

Reaction Control Nozzles--Reaction control nozzles will be mounted with the thrust axes normal to reference plane A within ± 2 degrees.

3.10.4 Assessment of Preferred System

3.10.4.1 Evaluation Summary

Table 3.10-1 summarizes the assessment of the preferred Voyager system against five evaluation criteria: probability of success, orbit characteristics, technical risk, spacecraft weight, and configuration design. The table shows that the specified mission success and orbit requirements can be met, that technical risk has been minimized by selecting all parts from a preferred parts list, that the spacecraft weight is within its allocation, and that the configuration design satisfies the major design criteria. These evaluations are briefly discussed in the following paragraphs and are explained in more detail in Volume B, Section 3.2.3.

3.10.4.2 Probability of Success

This criterion measures the ability of the system to achieve a successful mission on each launch attempt, i.e., to return the required data from the vicinity of Mars. Figure 3.10-6 is a plot that compares the estimated probability of success of the preferred system with that specified in JPL Document No. 45. Two estimates are shown: one for the combination of the launch vehicles and the Spacecraft Bus and propulsion, the other for the above systems plus the science payload. The latter curve is based on postulated instrument systems and includes a redundant TV camera. It is included to show the effect of the science payload reliability on overall mission success and to obtain estimates of overall mission

TABLE 3.10-1

EVALUATION SUMMARY

EVALUATION CRITERIA	PREFERRED SYSTEM	SPECIFICATION
Probability of Mission Success		
Total Mission (395 days)	0.25	None
Launch through 30 days in orbit	0.62	0.45
Launch through injection into Mars orbit	0.73	0.65
Launch through capsule separation	0.83	0.80
Data Acquisition and Recovery		
Orbit Periapsis per Period Coverage (Example)	2700 km per 18.3 hr Contiguous Repeat every 30 days	Contiguous Repeat 30-90 days
Occultation		
Sun (Days to Occultation)	55	> 30
Canopus		< 1-2 hrs per orbit
Earth	No data transmission degradation.	
Technical Risk	All parts from preferred list	Use minimum risk components
Spacecraft Weight (pounds)	5500	5500
Configuration Design	High rating against all design criteria	(See Sections 1.0 and 2.0)

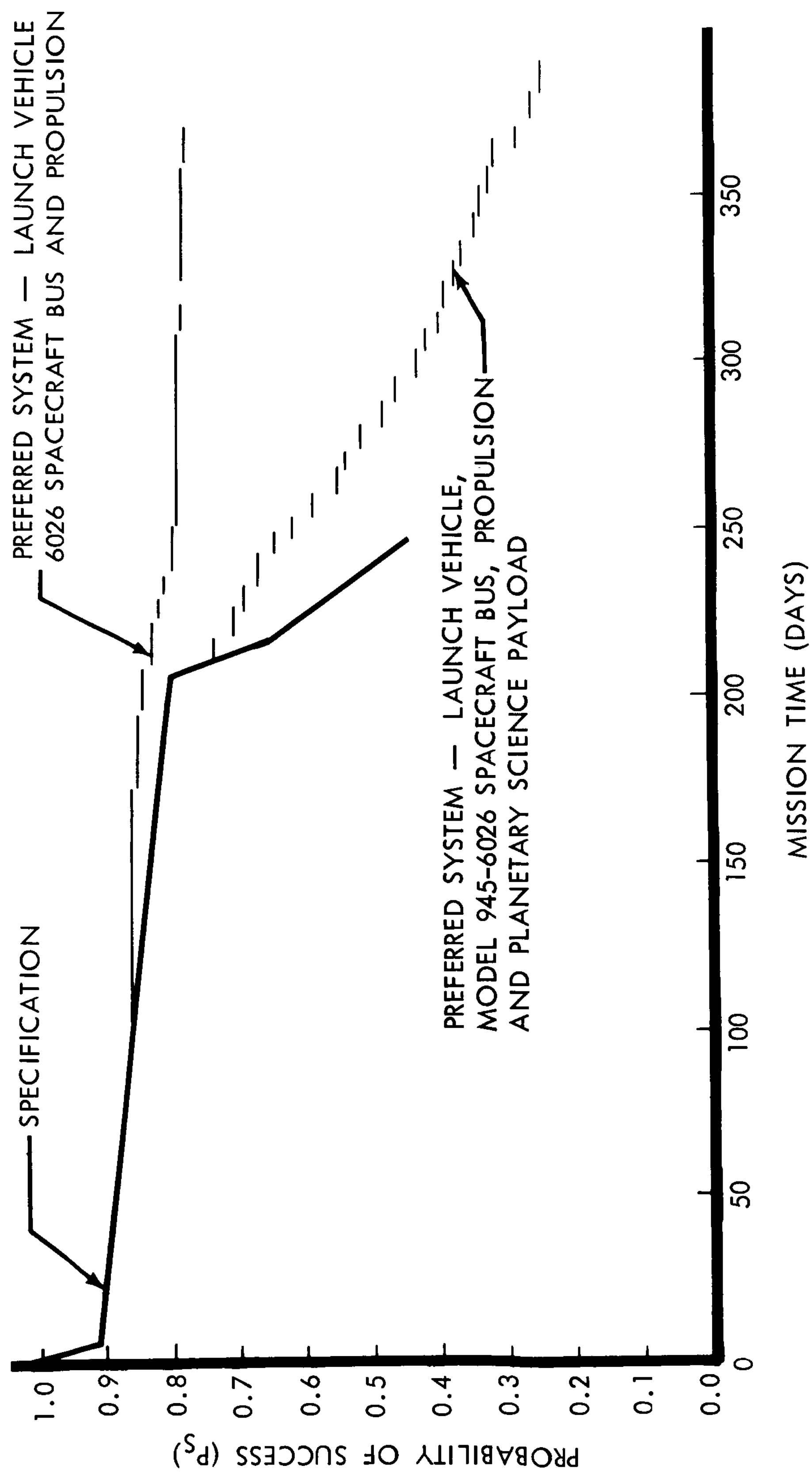


Figure 3.10-6: Probability Of Success — Preferred System

success for comparison with the mission specification.

3.10.4.3 Data Acquisition and Recovery

In the above mission-success analysis it was assumed that the required data would be obtained if the spacecraft were placed in and maintained in its prescribed orbit. The characteristics of an example orbit are shown on Table 3.10-1. The example orbit was selected from a range of available biologically safe orbits within the capability of the preferred spacecraft. (See Section 3.1).

3.10.4.4 Technical Risk

Technical risk measures the probability that the spacecraft components will be completely operational and meet their performance, reliability, and cost requirements by the spacecraft launch data. To maximize this factor a preferred parts list was assembled from existing preferred parts lists that includes only the most suitable parts for an interplanetary mission. All parts recommended for use in the Voyager system were then selected from this list.

3.10.4.5 Weight

As shown on Table 3.10-1 the entire spacecraft weight allocation was used. The weight "margin" available after initial system definition was used to maximize the probability of success as described in Section 3.2.3 of Volume B.

3.10.4.6 Configuration Design

The configuration design was rated in detail against the criteria of reliability, mass properties, views, and versatility. The evaluations were performed by breaking each of the criteria into several subcriteria, then rating the system against each of these subcriteria using a number-value from 1 to 10, 10 being best. This evaluation is reported in Section 3.2.3 of Volume B, and shows that the configuration chosen for the preferred system was the best of the alternatives considered, and is rated high against all criteria.

3.10.5 Maximization of Probability of Success (P_S)

3.10.5.1 Introduction

A continuing study to ensure maximum probability of mission success within system constraints such as contamination, weight, volume, power, and launch period was conducted throughout the Voyager Phase 1 contract.

A successful mission is defined as a mission during which a prescribed quantity and quality of data is received on Earth. To accomplish this, a system must successfully deliver data-gathering equipment to Mars and maintain it in the proper position for a prescribed period of time; the data must be gathered, sent back to Earth, and received.

The philosophy followed in this study was to maximize the probability of success considering the combined effect of all factors that contribute to success. The factors are those that determine whether each of the events of the previous paragraph can be completed. They are delineated below:

1) Reliability--the probability of no equipment failures that will prematurely terminate or significantly degrade the mission.
2) Performance--the combined probability that the system will perform within mission tolerances. This includes data quality and quantity tolerances.
3) Environment--the probability that possible ranges of radiation, charged-particle flux, meteorite flux, etc., can be successfully negotiated without mission failure.

3.10.5.2 Study Approach

The approach used is as follows:

1) All factors that contribute to mission success were identified to a level of detail commensurate with the Phase 1A design;
2) The combined effect of all the factors was then determined by a system simulation;
3) Defects in the system design were then corrected to the maximum extent possible by redesign;
4) The probability of success of the new design was then determined by the system simulation and the iteration was continued.

In correcting defects by redesign, considerations were made in the following order of priority:

1) Use of different subsystems or components;
2) Redesign of subsystems or components to increase reliability, or to provide a larger margin for performance or protection against the environment.
3) Addition of redundancy.

3.10.5.3 Probability of Success Maximization for Configuration 945-6026

The following discussion illustrates the method of maximization of probability of success (P_S) for Configuration Model 945-6026.

Figure 3.10.7 is a plot of probability of success versus mission time. It illustrates the specification of mission success capabilities as compared to the predicted capabilities on the preferred system for three iterations.

For the first iteration, the system was defined from a collection of independent subsystems and performance definitions without prior integration. Simulation and subsequent analysis of this system were accomplished and as shown on Figure 3.10.7, the predicted probability of success was significantly below the specifications even when the effect of failure of the scientific subsystems was not taken into account. Two sources were traced as causes of low success levels: low reliability and guidance error accumulation. The guidance error accumulation was rectified by improving the midcourse-correction-aim-point selection

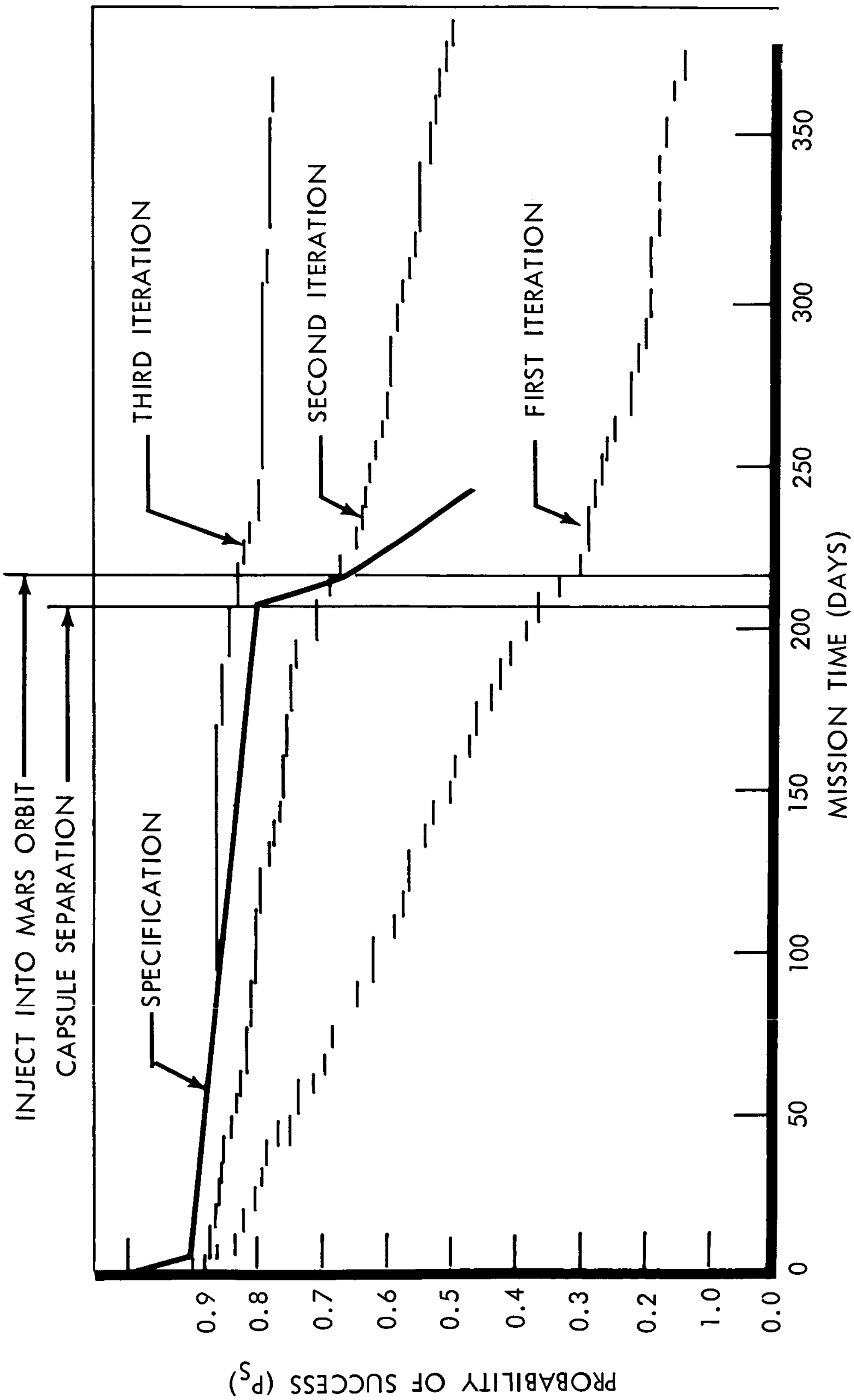


Figure 3.10-7: Probability Of Success For Launch Vehicle, Spacecraft Bus, And Propulsion

logic. An attempt was made to increase reliability through changes in subsystem components and limited incorporation of redundancy. These changes were incorporated to define a system for the second iteration.

As shown on Figure 3.10.7, this redifined system showed improved probability of success, but the objective of meeting the specification was still not attained. All system degradation in this sytem was traced to low reliability.

The process of defining the system for the third iteration was then reduced to maximizing reliability. This was accomplished by using more reliable components and adding redundancy within the maximum weight constraint of the spacecraft.

The method chosen for selecting components to be made redundant was one developed by Dr. Frank Proschan of Boeing Scientific Research Laboratories and described in "Mathematical Theory of Reliability" by Barlow and Proschan published in 1965 by John Wiley. In this method, maximum reliability is calculated first for the basic system. Progressive assignments of redundant components are then made by selecting the component that will result in the maximum increase in reliability per unit weight added. Redundant items are added until a weight constraint is reached or until the incremental improvement in P_S for a given weight addition is insignificant. The program output includes the sequence in which redundant components are added, the progressive increase in reliability, and the progressive increase in weight of the system. By choosing the first 20 redundant parts called for by the program, the Spacecraft Bus and propulsion system reliability was increased from 0.58 to 0.89 at the expense of about 100 pounds. These redundant components were incorporated to redefine the system for the third iteration.

It can be seen from Figure 3.10-7, that this third-iteration system exceeds the specifications at all mission times. The mission success definition used with the data of Figure 3.10-7 accounted for proper operation of the launch vehicle and the Spacecraft Bus and propulsion only. Figure 3.10-8 shows the effect of adding the planetary science payload. The highest line on Figure 3.10-8 is the same as that on Figure 3.10-6, and is valid for a definition in which mission success is considered attained if any science instrument and is able to return data. The second line from the top resulted from the assumption that the mission would be degraded an equal amount from the loss of any science subsystem component. The third line from the top resulted from the assumption that TV camera data was worth 90 percent of the mission and each of the remaining science instruments contribute only 1 percent each. The fourth line from the top resulted from the assumption that any science subsystem failure would completely fail the mission. These data illustrate a bracketing from the most optimistic to the most pessimistic definition of success. The data also shows that partial success evaluations are extremely sensitive to the judgments made on the value of the data obtained from each instrument.

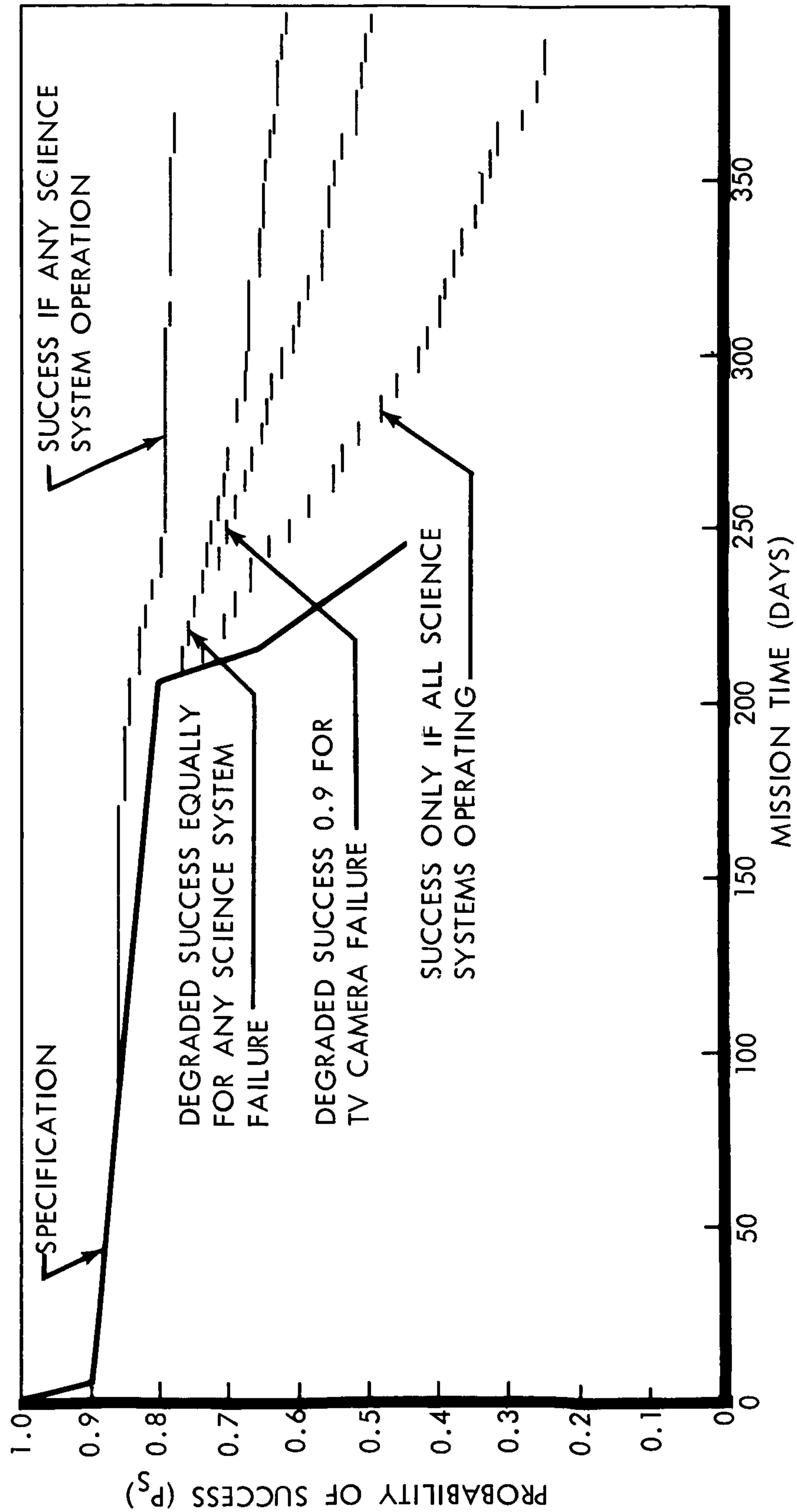


Figure 3.10-8: Partial Success Comparison

3.11 PLANETARY QUARANTINE

This section describes the techniques for complying with the planetary quarantine constraint: The probability that Mars is contaminated prior to the calendar year 2021 as a result of any single launch shall not be greater than 1 in 10,000. Included are the probability allocations assigned to the consideration of contamination by Centaur booster impact, capsule canister impact, Flight Capsule contributions, Flight Spacecraft accidental impact, propulsion systems exhaust products, and ejecta resulting from spacecraft meteoroid impact.

Sterilization of the orbit insertion and orbit trim propulsion systems and attitude control systems is specified. Further analysis is necessary before decontamination constraints are imposed on other Planetary Vehicle areas.

3.11.1 Applicable Documentation

The document applicable to this section is Boeing Document D2-82733-1, Planetary Quarantine Studies, July, 1965.

3.11.2 Functional Description

To reduce the probability of accidental impact of the Centaur booster case and the capsule sterilization canister, and of the Flight Spacecraft at encounter, the aim-points for the Planetary Vehicle trajectory have been biased so as to have a high probability of obtaining a nonimpact trajectory. The Centaur booster includes a retro capability, which, after separation, will be activated to ensure the probability of impact with Mars of 0.5×10^{-5} or less. With the Planetary Vehicle on a nonimpact trajectory at the time each of the capsule canister sections (forward and aft) are separated, the probability of their impacting Mars at encounter is 0.5×10^{-5}.

The majority of the contributions to the total probability of planet contamination caused by the Flight Capsule is assigned to capsule contractor. However, the separation mechanism is designed to a reliability factor of 0.999999+ and ensures the probability of malfunction resulting in violation of the biological barrier as no greater than 0.6×10^{-5}.

Biasing the aim-points results in a probability of accidental impact of the Flight Spacecraft at encounter of 1×10^{-5} or less. The probability of accidental impact of the spacecraft from orbit decay in less that 50 years will be no greater than 2×10^{-5}. Insertion command will not be given unless the resulting orbit satisfies this constraint. The final aim point selection is discussed in Section 2.4.

Contamination from propulsion system exhaust products will not exceed the probablity of 1×10^{-5}. The orbit insertion, orbit trim, and attitude control propulsion systems and components will be sterilized per JPL specification XSO-30275-TST-A to reduce the probability of contamination from emissions to 0.4×10^{-5}, 0.4×10^{-5} and 0.2×10^{-5}, respectively.

Analysis, as discussed in Section 3.3, Volume B, indicates that the biological contamination on the spacecraft external surfaces and protrusions subject to meteoroid impact should be reduced to meet the probability allocation of 2×10^{-5}. The analysis is based on the meteoroid environment presented in Section 2.2 of this volume. Considerable additional analytical work to evaluate all aspects of this problem more thoroughly should be performed before specific constraints are imposed on portions of the spacecraft other than propulsion systems.

3.12 VOYAGER FLIGHT-EQUIPMENT CLEANLINESS

3.12.1 Scope

Suggested procedures for cleanliness are presented in this section. Cleanliness is concerned with contamination that could adversely affect the mission. Two types of contamination, particulate and biological, are of special concern. Implications of control of the biological contamination have been presented in the previous section.

Particulate contamination has had many traceable and undesirable effects. It has been the cause of electrical systems failures where high-contact resistance could not be tolerated. It has been the significant cause of organic- and metallic-coating failures. The broad base of currently identified particulate contaminates has been responsible for lack of specification performance in parts fitted to microtolerances. Particulate contaminations carried aloft are suspected of causing both temporary and permanent malfunctions in the electrical and electronic systems, in various sensors, and in mechanical devices. Removal of these particulates is the object of the cleanliness process efforts. Procedures for obtaining cleanliness are effective to the extent that a proper organization for implementation and control has been established, and that people connected with any aspect of a cleanliness program have been properly selected, trained, and motivated. Cleanroom facilities will enable the spacecraft to undergo months of assembly and test, in a condition of total access, without acquiring an unacceptable particulate load from airborne sources. All particulate matter, as well as biological, will be reduced for a given class of cleanroom if the flow is vertical rather than horizontal.

3.12.2 Applicable Documentation

1) NPC 200-2, "Quality Program Provisions for Space Systems Contractors."
2) JPL Specification GMU-50387-GEN, "General Specification, Contamination Control of Flight Hardware, Spacecraft Control Flight Components." (Preliminary Draft)
3) Sandia Document SCTM 147-63(25), "Present Practices in the Verification of Cleanliness."
4) JPL Specification GMV-50004-PRS, "Process Specification, Cleaning and Contamination Control Procedures for Attitude-Control Gas-Actuator System."
5) JPL Specification GMV-50005-PRS, "Process Specification, Sampling Procedures for Fluid Contamination Control."
6) D2-100411-1, "Cleaning Reaction Control and Propulsion Subsystem Parts."
7) D2-100358-1, "Cleaning of Subassemblies and Parts Prior to Final Assembly in Class-100,000 Clean Rooms."
8) Federal Standard No. 209. Clean Room and Work Station Requirements, Controlled Environment.

3.12.3 Cleanliness Procedures

1) After a trial assembly of all mechanical interfaces, the Flight Spacecraft equipment shall be disassembled to its lowest practical level, cleaned, and reassembled in a Class-100,000 downflow facility (D2-100358-1). Equipment components that illustrate a lowest practical level of disassembly are such items as circuit cards and hermetically-sealed instruments. If, after receipt from another facility, such components are found to be externally contaminated, they will be cleaned externally to meet the conditions of cleanroom assembly. All components from subcontractors will be required to meet or better these assembly conditions. In some cases, the nature of the product will demand more stringent control; i.e., in the case of the gyros at least a Class-100 bench environment (Fed. Std. 209) will be required.

2) The liquid and gaseous elements of the reaction control and propulsion subsystems and associated components of these systems will be internally precleaned and then assembled in a Class-100 environment. Procedures for cleaning, inspection, packaging and handling of the subsystems and their components are contained in Boeing document D2-100411-1. Assembled systems will be post-checked in accordance with procedures and standards in D2-100411-1. Failure to meet the standards will result in disassembly and reprocessing.

3) All remaining Flight Spacecraft components will be cleaned, inspected, packaged, and shipped in accordance with procedures described in Boeing document D2-100358-1.

3.13 MAGNETICS

The influence of magnetics on the Voyager program is felt in the areas of: requirements, subsystems, system integration, design control, and facilities. It should be noted that magnetics in the sense discussed here is limited to slowly changing, quasi-static fields and specifically not a.c. or rf. This excludes radio frequency interference (RFI) and electrical interferences (EI) as used in aerospace design. It also leads to the most important underlying requirement for magnetic cleanliness, the magnetometer's measurement of interplanetary and Mars magnetic fields. These magnetometers measure about once a minute throughout the flight. Although this purpose is specifically stated to be a secondary objective of the Voyager program and tertiary objective of the 1971 Mission, the simplicity, lightness, and space experience with the magnetometers suggests the continuance of magnetic measurements.

The magnetic requirements are illustrated in Table 3.13-1. They are determined by scientific interest in magnetic fields, the possibility of destabilization of the vehicle due to field interaction, and by the specifications imposed in the contract. In the area of scientific interest, the Mariner IV detection of no magnetic dipole of Mars may reduce somewhat the concern for magnetic measurements as a scientific mission. However, interplanetary fields change with time as well as position and corroborating data about the Mars magnetic field is still very valuable. An examination was made of magnetic torques that might upset the vehicle or use up attitude control propellant. This torque is caused by the residual magnetic dipole of the spacecraft in the ambient field. Considering the spacecraft dipole moment to be 104 dyne-cm/gauss and the ambient field to be 100 gamma,

TABLE 3.13-1

MAGNETIC REQUIREMENTS		
Scientific Interest		
Measure Interplanetary Fields		0 to 20 gamma
Measure Mars Field (in orbit)		20 (to 5000) gamma
Vehicle Stabilization		
Field Interaction	Maximum $< 10^{-6}$ pound-feet	
Specifications		
Materials	Nonmagnetic whenever possible, all evaluated	
Magnitudes	$<$ 1 gamma at 3 times "assembly" dimension	
Stability	Change $<$ x 10 for 25 oersted on spacecraft	
	Change $<$ x 10 for 100 oersted on assemblies/ components	
Stray Fields	Change $<$ x 1 gamma at 2 feet; 5 gamma at 1 meter	
Design Verification	Map perm and current fields of spacecraft	

the maximum torque, $T = M \times B \approx 10^{-6}$ foot-pounds. This torque is trivial in vehicle stabilization.

The magnetic environment in Figure 3.13-1 shows on a single logarithmic scale the relationships between gauss, gamma, and oersted for the nine decades (or orders of magnitude) pertinent to Voyager. It starts at the surface of Mars, now known to have no magnetic dipole and proceeds along the irregular line indicating the uncertainty in the field both for Mars and interplanetary space. The interplanetary field then blends into the well-defined field of the Earth. The scale of sizes of the spacecraft and its subsystems are indicated also because the specifications for the perming and deperming fields and the detectable changes in the fields are related to these dimensions. And, finally, the range and precision of the magnetometer for the Mariner IV, 0 to 350 $\pm$ 0.5 gamma is shown in solid lines. The potential range to 20,000 gamma is shown dotted.

Table 3.13-2 lists the spacecraft subsystems in the order of the Phase II Voyager program breakdown structure. Magnetic components have been identified for these subsystems. An estimate of the expected field strength at three times the assembly dimension is listed. Four of the systems probably will surpass the less than 1 gamma specification. Specifically, an examination of these systems now indicates unavoidable excessive local fields (at three times dimension of "assembly") will be caused by the traveling wave (x 10 too great), the attitude control motor (x 3) and the autopilot and CC&S electronics (x 5). However, the fields

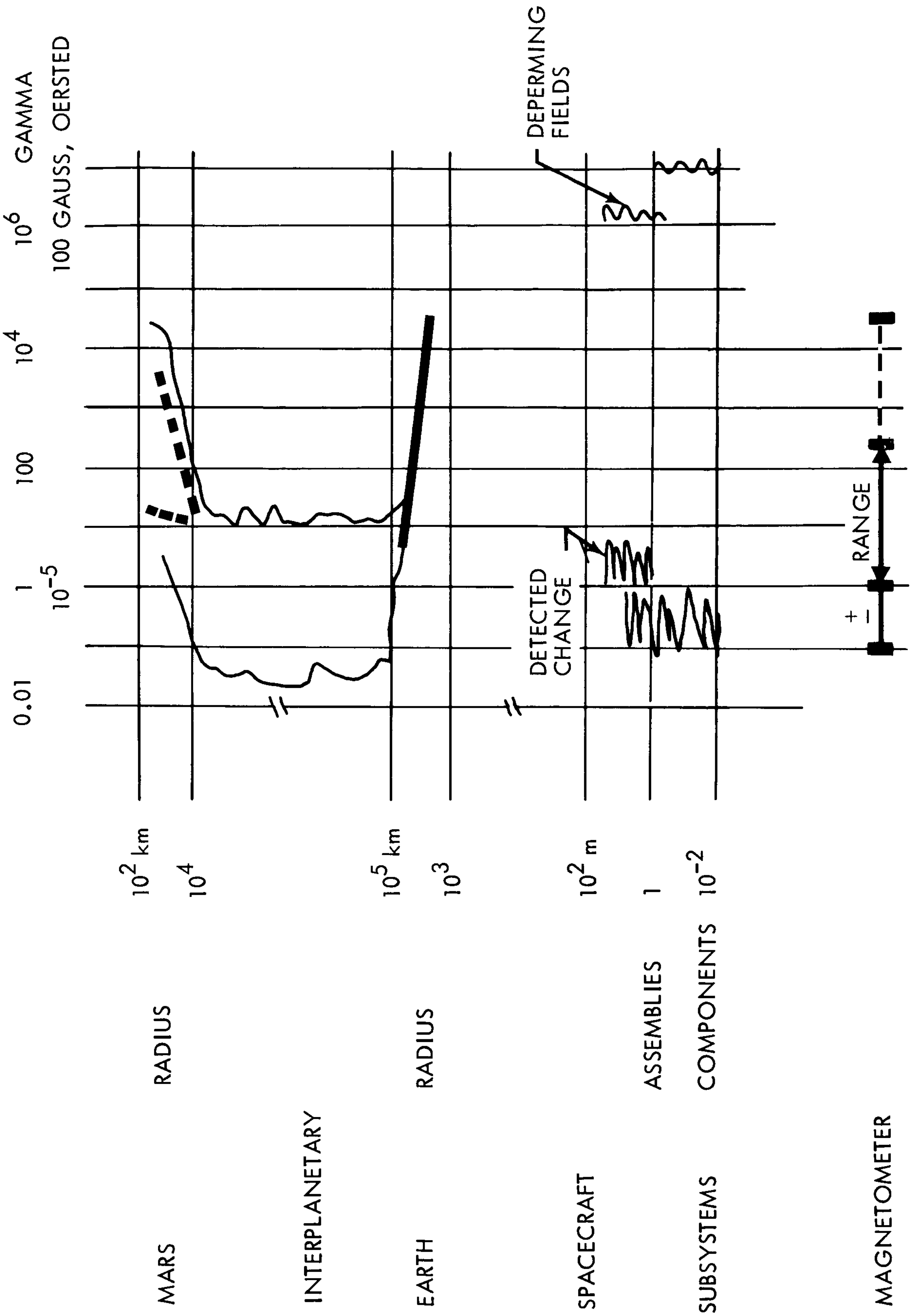


Figure 3. 13-1: Magnetic Environment

TABLE 3.13-2

SUBSYSTEMS		
Subsystems	Critical Magnetic Components	Field at (Assy X 3) Gamma
Telecommunications	Traveling-Wave-Tube Power Converter	10
Attitude Reference	Gyro, Accelerometer, Power	3
Autopilot	Electronics	3
Reaction Control	Valve Actuators	< 1
CC&S	Magnetic Memory, Nickel Ribbons	5
Electrical Power	Solar Panel, Power Conditioning	≈ 1
Structures	None	0
Mechanisms	Actuators, Motors, Bearings	< 1
Temperature Control	None (provided that special louver actuators are used)	< 1
Pyrotechnics	None	< 1
Cables & Tubing	Loops, leakage	
Midcourse Propulsion	Thrust Chamber, Solenoids	< 1
Injection Propulsion	None	< 1
Science Payload	Actuators, Motors	< 1

due to these items are expected to be characteristic of higher order multipoles (quadrpoles or higher) so that their strength will diminish with distance at rates exceeding that associated with a dipole (inverse cube of distance). For that reason, these systems are not expected to disturb the magnetometer measurements at the magnetometer's position. This is indicated in Table 3.13-3 where Voyager magnetic properties are compared with those of Mariner IV.

To ensure that the subsystems and systems will measure up to the requirements and performances stated above, a magnetics control plan includes four essential elements.

1) Nonmagnetic materials are specified throughout. Whenever the design cannot avoid magnetic materials, the requirements for them must be justified in detail and specific project authorization must be obtained.
2) The proposed designs are evaluated by superimposing the fields produced by the several subsystems by vector-addition of the fields. This vector summing of the fields would be exact except for the presence of ferromagnetic material as components of some of the subsystems.
3) Subsystems are tested as necessary to verify design evaluation. All subsystems will be depermed through application of peak fields of 200 oersteds at 60 cps. The test procedure requires subsequent application of a 100-oersted magnetizing force field, and mapping of the resultant magnetic field to demonstrate that residual fields are less than 1 gamma at 2 meters.

TABLE 3.13-3

SPACECRAFT INTEGRATION

Magnetic Environment At Magnetometer	Voyager	Mariner IV
Estimated Total Field	10 γ	30 γ
Voyager Specification RQMT	undefined	
Magnetic-Field Stability	± 1 γ	± 2 γ

Field Stability Rather Than Field Magnitude Is The Critical Parameter In Magnetic Mapping of Space.

Anticipated Magnetometer Measurement Capability:

± 350 γ	±.1% of range
± 20,000 γ	±.1% of range

4) The spacecraft will be tested by first applying a 25-oersted magnetizing force fields. Mapping will be performed to demonstrate that residual fields are less than 1 gamma at 8 meters. The mapping will then be repeated while the spacecraft is operated under rated power.

To verify the magnetic specifications on the spacecraft and the subsystems, they must be tested in a facility capable of mapping, perming, and deperming. (See Table 3.13-4).

A Helmholtz or Braumbek orthogonal coil system is required that would maintain a uniform (± 1 percent of center field at 25-foot radius), low ($\pm 1\ \gamma$) within 2-foot radius, and stable ($\pm 1\ \gamma$) magnetic field throughout a 50-foot sphere. In addition, a separate set of perm-deperming coils capable of providing a 150 oersted (peak) field at 60 cps for several seconds would be required.

An alternative to the 50-foot facility may be considered if the deployable members are not magnetically tested with the spacecraft body. Solar panels and booms contribution to magnetic fields at the magnetometer is expected to be less than 1 γ. Individual magnetic verification of the deployable assemblies coupled with analytical predictions could produce an acceptable magnetic verification program for the measurements and reduce (to one-half in size) the facility requirement.

TABLE 3.13-4

MAGNETIC FACILITIES

Assemblies	Internal Size		Type	Specification
8" x 16" x 24"	4' cube		Orthogonal Helmholtz Coil	Range 0 - Ambient ± 1
6" x 103" x 120" (Solar Panel)	12' cube			Gradient - ± 1 of test level within cube
Spacecraft				Stability ± 1
41' x 41' x 5'	50' cube		Plus Remote Ambient Field Controller	
Existing	16' x 16' x 50'	Goddard		
	22' cube	Goddard		
	22' cube	AMES		
	30' cube	STL		
	20' x 20' x 40'	Philco		
	22' sphere	Malibu, Calif.		
	44' cube	Goddard		
Perm and Deperm	12' coil	200 oersted at 60 cycles for 60 seconds		
	50' coil	100 oersted at 60 cycles for 60 seconds		

3.14 SPACE RADIATION EFFECTS ON THE VOYAGER SYSTEM

3.14.1 Scope

This section is concerned with the radiation effects on components and assemblies and the types of system responses or failures that are likely to ensue from space radiation predicted for the Voyager mission. A large amount of data already exists on the basic radiation responses of electronics and materials intended for use in Voyager. More data is being obtained on specific devices and materials under specific test conditions relatable to the Voyager radiation criteria. Also, a broad competence in radiation vulnerability and system hardening has been developed at Boeing over the years, so that, with the existing data, an integrated design can be obtained in which radiation responses are accounted for and do not lead to unacceptably low reliability values. The radiation effects likely to be most important to the Voyager system are surface and ionization effects rather than bulk damage to electronics. These surface phenomena have only recently come under study, so there is yet much component data to be obtained. However, enough data exists to indicate that high-gain low current transistors and power supply solar cells assemblies, diodes, and transistors may be candidates for scrutiny and for component selection on the basis of radiation response. Preliminary system analysis of the subsystem block diagrams indicates that the radiation effects problems are soluble because they are being attacked at the system level early in the design.

3.14.2 Applicable Documents

1) JPL Inter-Office Memo (with attachment) D-62-65 (Boeing No. 5-7680-LA-119).

2) MIT Lincoln Labs, 5 Feb. 1965, Group Report 1965-11 by A. Stanley. "Space Radiation Effects on High Gain Low Current Silicon Planar Transistors."

3) Boeing Document D2-90412. "A Study of Solar Cell Space System Radiation Vulnerability," by R. Brown.

4) Boeing Document D2-90463. "Radiation Effects on Semiconductor Surfaces -- A Survey of Existing Evidence and Proposed Experiments" by J. F. Aschner, March 3, 1964.

5) Boeing Document D2-90570. "Proton and Electron Permanent Damage in Silicon Semiconductor Devices," by R. R. Brown, September 23, 1964.

6) Boeing Document D2-36222-1. "Experimental Results of Radiation Damage in Solar Cells," by R. J. Tallent.

7) Boeing Document D2-36359-1. "Space Radiation Tests on Reflecting Surfaces -- Final Report (Contr. JPL-950998)," R. Gillette, R. Brown, 2 June 1965.

3.14.3 Semiconductor Device Responses

3.14.3.1 Bulk Damage

Geomagnetically trapped protons and electrons, solar-event protons, and alpha particles can cause permanent displacement damage and both permanent and temporary surface ionization damage to transistors. Because the high-energy proton and electron environment levels for the Voyager missions are much lower than the values shown by experiment to be needed for significant bulk damage in many devices, it is felt that these effects will not present a serious system problem.

3.14.3.2 Ionization and Surface Effects

For the Voyager, ionization and surface effects are likely to be important mechanisms. Ionization from charged particles, including those having energies below the displacement threshold, can cause temporary loss of transistor gain, increase in leakage current, and reduction in breakdown voltage. Failure to recognize and account for these surface ionization effects could lead to malfunctions.

Data indicate that special surface treatments and manufacturing techniques can reduce these effects and therefore selection and screening of devices are possible for specific applications (Ref. MIT 1965-11). For example, leakage current of the Sperry 164N2 transistor increased from few nanoamperes (at about 10^9 electrons/cm^2 -sec) to microamperes after irradiation with 10^{13} electrons/cm^2 total fluence. This is a worst case and most other devices did not degrade so much. The 2N930, which is proposed for use in the Voyager Autopilot System, changed its leakage current only a few nanoamperes under the same irradiation conditions.

Experimental data show that NPN silicon transistors with low initial leakage current (unirradiated) tend to respond to dose rate and recover quickly, whereas those with high initial leakage currents tend to respond to total dose and to recover slowly (ref. D2-90463). All these facts lead to device selection criteria for space applications.

Surface ionization damage affects devices in the small-effect low-dose part of the damage curve. The Voyager mission will expose the electronics to sufficient particle fluence and flux so that account must be taken of the effects in the design, especially for extended Mars orbiting missions.

3.14.3.3 Solar Cells

Solar cells are discussed in detail elsewhere in the document. However, they are known to be radiation sensitive, and so are mentioned here as well. Solar cell assemblies will be affected by both the solar wind (low energy) protons and by solar ultraviolet radiation.

In general, N/P silicon cells selected for the preferred design are more radiation resistant than P/N, as measured by changes in short circuit current and open circuit voltage.

Much of the loss in solar-cell output in the devices tested in the past has been shown to be due to cover glass transmission change (D2-36222-1 and D2-90412). SiO_2 glass covers can be expected to darken appreciably during the flight to Mars, but experimental data indicate that quartz covers for the solar cells will not be degraded by the assumed fluences.

3.14.3.4 "Radiation-Preferred" Electronics Parts List

A detailed evaluation of electronic components on the JPL-preferred parts list ZPP-2061-PPL-F has been made, and the results given in a Boeing memo (2-7861-20-400, 3 May 1965), "Identification of Radiation-Preferred Electronics." In addition to discrete transistors and diodes, integrated circuits, SCR's, Zener diodes, FET's, capacitors, resistors, and insulating materials are assessed. These data include safety margins, or ratios of damage threshold levels to mission environment levels, for the Voyager. These results have been included as an attachment to JPL Inter-Office Memo D-62-65 (Boeing No. 5-7680-LA-119).

3.14.4 Voyager System Materials

Materials are discussed throughout the entire document, but a few summary remarks are made here on radiation effects, because these effects present system analysis and design problems.

Insulating materials or dielectrics in electronic systems may be degraded by radiation and lead to failures. Teflon, PVC, and mylar are examples of radiation-sensitive materials, as determined by laboratory tests.

Other materials problems include thermal control surface materials, solar-cell reflector surfaces, sealants, and other organics that are exposed.

Boeing research studies, Lunar Orbiter engineering data, and Mariner IV experience have indicated that the performance of thermal control coatings used in the Mars mission will be quite dependent on the combined effects of particulate (proton and electron) and electromagnetic (ultra-violet) radiation on absorptance and emittance properties of the coatings.

The radiation effects on thermal control coatings arise from both ionization and lattice defect production on the material. Color centers are formed that act as light-absorption sites. A Beoing-developed barrier-layer anodic coating on aluminum has been shown far more resistant (i.e., stable) than the others tested, which included vapor-deposited aluminum, bright aluminum, $ZnO/KSiO_3$, and ZnO/LTV-602 (Ref. D2-36359-1). This barrier-layer anodic coating will function in the radiation environment expected in a Mars mission. Figure 3.14-1 shows some of the test results. Ultraviolet irradiation produces effects similar to particle radiation, and preliminary studies indicate that these effects act synergistically with the particulate radiation effects. Presently, studies are beginning at Boeing on contract NAS 5-9650, to study the combined environmental effects of particles, UV, vacuum, and temperature on selected thermal control coatings. Results of these studies will lead to design data to ensure high reliability of the Voyager thermal control system.

3.14.5 System Responses

3.14.5.1 Power Supplies

All electronic systems in the Voyager craft use power supplies with diodes and transistors, and ultimately depend on solar cells for power. Responses of these devices can lead to lowered performance of the power systems.

The Mariner vehicle's problem of acquiring Canopus in its star tracker and the encoder problems encountered shortly after launch were due to Van Allen belt radiation-induced effects in the optical or electronic systems, or both. The trapped radiation dose rates (10-100 rad/hr) are high enough to give ionization and surface effects in semiconductors, which can lead to reduction of transistor gain and increased diode leakage currents. A circuit analysis of the Marine IV star seeker system was made by Boeing to assess the radiation vulnerability of this type of equipment (also used on Lunar Orbiter). It was found that a power supply

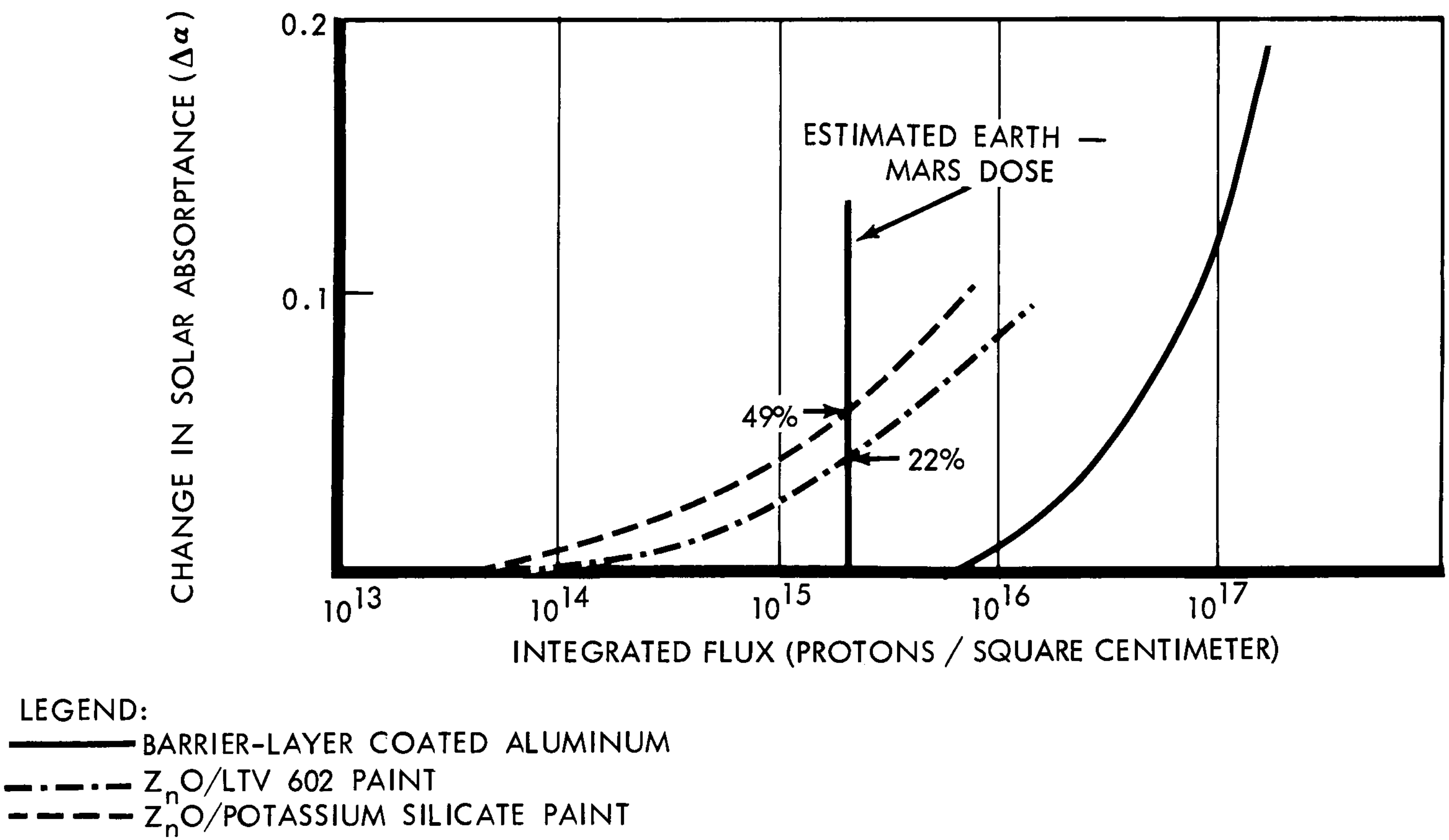


Figure 3.14-1: Changes In Solar Absorptance Produced By Low-Energy Protons

diode type is being used that has a high reverse leakage current under radiation. The response to trapped radiation dose rates would be high enough to decrease power supply voltages to the Mariner IV electronics. Another of the potential weak links in the system was found to be surface-induced gain changes in certain amplifying transistors.

3.14.5.2 Central Computer and Sequencer

Radiation-induced permanent and temporary changes in low current gain and reverse leakage current in the semiconductor elements will reduce reliability and could cause malfunctions. A preliminary analysis of the preferred design has not indicated any radiation-soft blocks. However, as the detailed design evolves, the constraints imposed by radiation hardening requirements will be applied, and trades will be involved. Device selection criteria will include low radiation response. Block diagram outputs and inputs will be considered to be sure that radiation-induced spurious signals do not exceed allowed ranges for proper performance. For example, if some transistor gain decreases by 10 percent, the circuit will be designed so that it still functions and produces an output signal close enough to the correct one so that the system will function. These are well known hardening techniques from missile technology. Each circuit or system requires its own detailed analysis in terms of radiation vulnerability. Such an analysis involves consideration of the environment, the mission, system failure modes, component responses, and component interconnections. Many volumes of test data have been accumulated at Boeing on component and circuit response to radiation from contracts and company-sponsored research. This data and circuit design information is available for integrating the radiation vulnerability specifications into the original design of guidance and communication electronics for Voyager.

3.14.5.3 Attitude Reference and Autopilot

The same general considerations apply to this subsystem as to the central computer and sequencer already discussed. The use of linear amplifying circuits (some are integrated circuits) implies careful selection to avoid gain degradation, and to obtain well-balanced difference circuits for common-mode rejection of radiation effects. It also means design consideration must be given to reducing effects of power supply voltage changes, by feedback, differencing, or other means. An integrated approach is made to the circuit design, in which radiation response is considered as a system constraint along with the other factors.

A potential problem area for a star tracker is the increase in phototube dark current, due to radiation interactions on the photocathode. Tests have indicated that luminescence in the glass components will lead to spurious currents in a star tracker input circuit for up to 15 minutes after irradiation at a 30 rad/hr rate. Careful balancing of components and design to reject common mode noise can reduce the system response. This type of problem should occur only during near-Earth or near-Mars maneuvering. It may be possible to program the Canopus-seeking operation at a time after the radiation belts are traversed.

Similar considerations hold for the sun-seeker subsystem, where the photocell response to radiation becomes a part of the circuit design. Much CdS photocell data exists for this analysis.

Mechanical instruments, such as reference gyros, are relatively hard to radiation at the dose levels to be encountered on the Voyager mission. However, gyro power supplies may use high-power diodes or transistors whose performance degrades, and whose outputs fall below acceptable limits. By device selection, and or by use of switching-type power-regulator circuits, these difficulties are overcome. The organic materials in a gyro are normally well enough shielded to avoid serious degradation of properties, but a relatively simple gyro and radiation-resistant materials were selected for the preferred design to minimize the effects.

3.14.5.4 Telecommunications

The power supply is probably the radiation-weak link in the communication subsystem, but linear and power devices of the transmitter also need consideration. Device selection and hardened circuit design techniques are applicable, so that no insoluble problem is anticipated.

ERRATA

The Boeing Company Document D2-82709-1
Voyager Spacecraft System Final Technical Report

Volume A

"Preferred Design for Flight Spacecraft and Hardware Subsystems"

Page No.	Paragraph, Table, or Figure No.	
		PART I
17	Summary	Delete: "6) The trade-off between . . . inherently" Add: 6) The trade-off between proven instruments versus new and inherently simpler instruments. 7) Determination of the degree and type of redundancy,for example, using two identical instruments of two different designs. 8) The effect of the solid engine exhaust on the structure and solar panel temperature. 9) Accommodating the length of the orbit insertion engine. 10) Selection of installation technique for the equipment packages. 11) Selection of the thrust vector control technique. 12) Effect of heat soak sterilization on equipment. These problems are the key technical considerations in developing the preferred design. The subsystem of the Boeing team's spacecraft provide a conservative and highly reliable design. No state-of-the-art advances are required to meet the design criteria for any subsystem.

ERRATA (Continued)

Volume A

Page No.	Paragraph, Table, or Figure No.	
✓2-3	Para. 2.1.2.1	First paragraph, last line: Change: "low-gain" to "ascent"
✓2-9	Para. 2.1.3	Line 2. Change: "A low-gain S-band antenna capable of providing uniform coverage during the near-Earth ..." to: "A low-gain S-band antenna capable of providing uniform coverage for telemetry during the cruise portion ..." Add: 4) An ascent antenna system for telemetry during launch and early cruise."
✓2-11	Para. 2.1.3	Last paragraph: Change: "The spacecraft shall be instrumented to provide two-way doppler and telemetry data for ranging." to: "The spacecraft shall be instrumented to provide ranging capability."
✓2-13	Para. 2.1.3.3	Add sentence to last paragraph: "Design characteristics and restraints for these subsections given in Table 2-1." Add: "Table 2-1" (attached)
2-14	Para 2.1.3.3	Delete: First four paragraphs on page.
2-16	Para. 2.1.3.4	Line 2 from bottom of page: Change: "135°F" to "120°F"

ERRATA (Continued)

Volume A

Page No.	Paragraph, Table, or Figure No.	
3-13	Table 3.1-2	In Line 3.5, Column: "Possible number of bits/sec" Change: 84,000 to "84,000[2]" Add reference: "2. Preferred design provides 48,000 bits/second."
3-42	Figure 3.1-22	Change: r_o = 3.360 km to: "r_o = 3,360 km" (in legend)
3-67	Para. 3.1.5	Line 3 from bottom of page: Delete: "The DSN tracking limits ... of 18 km^2/sec^2."
	Figure 3.1-43	Replace with revised figure attached. Add page number: "78A."
3-87	Table 3.3-2	Spacecraft Telecommunications, Column S/C Bus Change from "227" to "207"; Change from "166" to "186."
3-109	Figure 3.5-5	Change 227 R to "238 R," Change 64.5 to "59."
3-110	Para. 3.5.3.1	Delete: "1) RF coupler ... shroud." Add: "1) RF coupler and shroud antenna."
3-110	Para. 3.5.3.3	Delete: 2) Spacecraft S-band transmission ... 2048. Delete: The latter interface ... separation occurs. Add: "2) Spacecraft S-band transmission via the shroud antenna located below Saturn Station 2048. The coax cabling for this antenna includes an RF coupler at the interface (Saturn Station 2048.)"

ERRATA (Continued)

Volume A

Page No.	Paragraph, Table, or Figure No.	
3-111	Figure 3.5-6	Delete: "Ascent Antenna (S-Band)" (and callout) Change: "Parasitic Antenna" to "Shroud Antenna."
3.114	Para. 3.5.6.2	Delete: Last two sentences from "The data-encoder ... pulse train." Add: "Spacecraft and capsule data in Telemetry Mode 1 at 22-2/9 bits per second are sent to the Centaur via an isolation amplifier."
3.117	Para. 3.6.2	Change: Document No. D2-23834-1, Rev. A, to D2-82724-3, Rev. B.
3.127	Para. 3.7.2.2	4) Change "23 bits" to "26 bits"
3-149	Para. 3.8.2	Line 6 from bottom. Change: "Accelerometer Null Bias - 100 x ... 2.3%" to "Accelerometer Null Bias = $100 \times \frac{1.5 \times 10^{-4} g}{0.013\ g}$ = 1.15%."
3-159	Figure 3.9-2	Line Spacecraft Telecommunications, Column 3.0. "Acquisition": Change: "c) from receive via low-power launch exciter detect and send to CC&S command signals from earth" to: "c) Receive, detect and send to CC&S command signals from Earth."
3-257	Para. 3.10.3.2	Line 5: Change: 87 degrees to "183 degrees"
3-261	Para. 3.10.4.5	Change last line on page from; "Section 3.2.3 of Volume B" to "Section 3.10.5, Volume A"